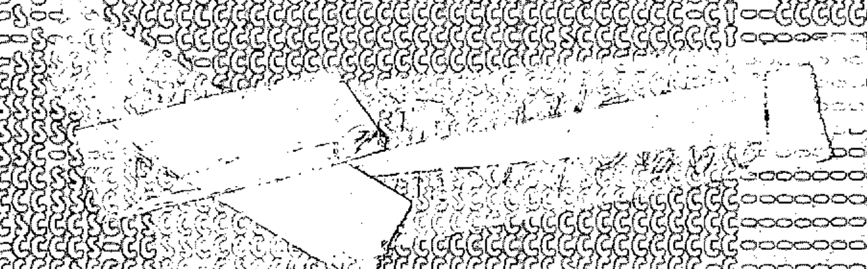

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 3: DESIGN/COST TRADEOFF STUDIES

- Appendix D: EOS Configuration Design Data
- Part 1: Spacecraft Configuration

(NASA-CR-143666) EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY. REPORT NO. 3: DESIGN/COST TRADEOFF STUDIES. APPENDIX D: EOS CONFIGURATION DESIGN DATA. PART 1: SPACECRAFT CONFIGURATION (Grumman

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EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 3: DESIGN/COST TRADEOFF STUDIES

- Appendix D: EOS Configuration Design Data
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Prepared For

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September 1974

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1. SPACECRAFT CONFIGURATION

1.1 Spacecraft and Instrument Support

1.1.1 Instrument Section Requirements

The structure subsystem shall provide the mounting and support for the instrument compliments, Instrument Mission Peculiar modules, antennas and solar arrays for the EOS missions shown in Table 1.1.1-1. The instrument section structure shall meet the design requirements specified in Paragraph 1.1.1.3.

1.1.1.1 Spacecraft Requirements

The structure subsystem shall provide for:

- o Common core structure for EOS missions A, A', B and C
- o (3) S/S equipment modules (EPS, ACS and Communications & Data Handling Modules)
  - \- Resupply capability
- o Orbit adjust/RCS/orbit transfer systems as required
  - \- Resupply capability
- o Mounting provisions to mate with a Delta 2910 or Titan III B launch vehicle
- o Shuttle launch and retrieval
- o Provisions for resupply of instruments and mission peculiars
- o Design requirements specified in Paragraph 1.1.1.3.

1.1.1.3 Design Requirements

1.1.1.3.1 Spacecraft Structure

1.1.1.3.1.1 Structural Rigidity

To avoid dynamic coupling between the low frequency launch vehicle and spacecraft modes, the launch vehicle contractors have recommended the minimum frequency criteria given in Table 1.1.1-2. These imposed criteria are for the spacecraft constrained at the Spacecraft/Launch Vehicle interface.

TABLE 1.1.1-1

INSTRUMENT SECTION REQUIREMENTS

EOS MISSION	SPACECRAFT PAYLOAD	INSTRUMENT MISSION PECULIARS	ANTENNAS	SOLAR ARRAY	LAUNCH VEHICLE
A	(1) MSS (1) TM DCS	(1) 11 x 25 x 32 inch Recorder Module (1) 14 x 36 x 36 inch IMP Module	(1) X-Band Steerable (1) X-Band Shaped Beam	155 Sq. Ft 516 Watts	Delta 2910
A'	(1) MSS (1) HRPI DCS	Same as A	Same as A	Same as A	Delta 2910
B	(1) TM (1) HRPI DCS	(1) 22 x 30 x 36 inch Recorder Module (1) 14 x 36 x 36 inch IMP Module	Same as A	Same as A	Delta 3910
C	(2) TM (1) HRPI (1) SAR DCS	Same as B	Same as A	230 Sq. Ft. 766 Watts	Titan III B

TABLE 1.1.1-2 MINIMUM FREQUENCY CRITERIA

LAUNCH VEHICLE	Minimum Frequency, Hz		Reference
	Longitudinal	Lateral	
Delta	35	15	1
Titan III B/NUS	20	10	2
Shuttle	N.D.	N.D.	3

N.D. = Not defined

1.1.1.3.1.2 Load Factors

The load factors (limit and ultimate) specified by the Launch Vehicle contractor are given in Tables 1.1.1-3, - 4 and 5 for Delta, Titan III B/NUS and Shuttle respectively. Figure 1.1.1-1 shows the Spacecraft coordinate system sign convention. These load factors were obtained from References 1 to 3.

1.1.1.3.2 Subsystem Module Structure

1.1.1.3.2.1 Structural Rigidity

Reference 4 specifies that the first lateral and vertical natural frequencies of a fully loaded module, constrained at the four corners, shall be greater than 60 H_z.

1.1.1.3.2.2 Steady-State Acceleration

Reference 4 specifies that the module structure shall be designed for a maximum steady-state acceleration of 25 g longitudinal and 15 g lateral.

REFERENCES

1. Delta Spacecraft Design Restraints, McDonnell Douglas Document DAC-61687, dated November 1973. In addition, McDonnell Douglas letter A3-110MJS-74-58, Mr. M. J. Schmitt to Mr. J. Marino, dated 31 May 1974.

Table 1.1.1-3 Load Factors Delta 2910 and Weight Constrained TITAN IIIB

CONDITION	LIMIT LOAD FACTORS		ULTIMATE LOAD FACTORS (1)	
	LONGITUDINAL X	LATERAL Y or Z	LONGITUDINAL X	LATERAL Y or Z
LIFT-OFF	+2.9 -1.0	2.0	+4.35 -1.5	3.0
MAIN ENGINE CUTOFF	+12.3	0.65	+18.45	1.0

Table 1.1.1-4 Load Factors TITAN IIIB/NUS

CONDITION	LIMIT LOAD FACTORS		ULTIMATE LOAD FACTORS (1)	
	LONGITUDINAL X	LATERAL X or Z	LONGITUDINAL X	LATERAL Y or Z
LIFT-OFF	+2.3 -0.8	2.0	+3.45 -1.2	3.0
STAGE I SHUTDOWN (depletion)	+8.2 -2.5	1.5	+12.3 -3.75	2.25
STAGE I SHUTDOWN (command)	+10.8 -2.0	1.5	+16.2 -3.0	2.25

NOTES: 1. Limit load factor times 1.5
2. Load factor carries the sign of the externally applied load.
3. Includes both steady state and dynamic conditions.

TABLE 1.1.1-5 PAYLOAD BAY LOAD FACTORS SHUTTLE (4)

CONDITION	LIMIT LOAD FACTORS			ULTIMATE LOAD FACTORS (1)		
	DIRECTION (2)			DIRECTION (2)		
	X	Y	Z	X	Y	Z
LIFT-OFF (3)	+1.7 ± 0.6	±0.3	+0.8 +0.2	+2.55 ± 0.9	±0.45	+1.2 +0.3
HIGH Q BOOST	+1.9	±0.2	-0.2 +0.5	+2.85	±0.3	-0.3 +0.75
BOOSTER END BURN	+3.0 ± 0.3	±0.2	+0.4	+4.5 ± 0.45	±0.3	+0.6
ORBITER END BURN	+3.0 ± 0.3	±0.2	+0.5	+4.5 ± 0.45	±0.3	+0.75
SPACE OPERATIONS	+0.2 -0.1	±0.1	±0.1	+0.3 -0.15	±0.15	±0.15
ENTRY	±0.25	±0.5	-3.0 +1.0	±0.38	±0.75	-4.5 +1.5
SUBSONIC MANEUVERING	±0.25	±0.5	-2.5 +1.0	±0.38	±0.75	-3.75 +1.5
LANDING AND BRAKING	±1.5	±1.5	-2.5	±2.25	±2.25	-3.75
CRASH (5)	—	—	—	-9.5 +1.5	±1.5	-4.5 +2.0

NOTES:
1. Limit load factor times 1.5 except for crash.
2. Positive X, Y, Z direction equal forward, right and down.
3. These factors include dynamic transient load factors.
4. These factors do not include dynamic response of the payload.
5. Crash load factors are ultimate and only used to design local payload support. fittings and attachments. The specified load factors shall apply separately.

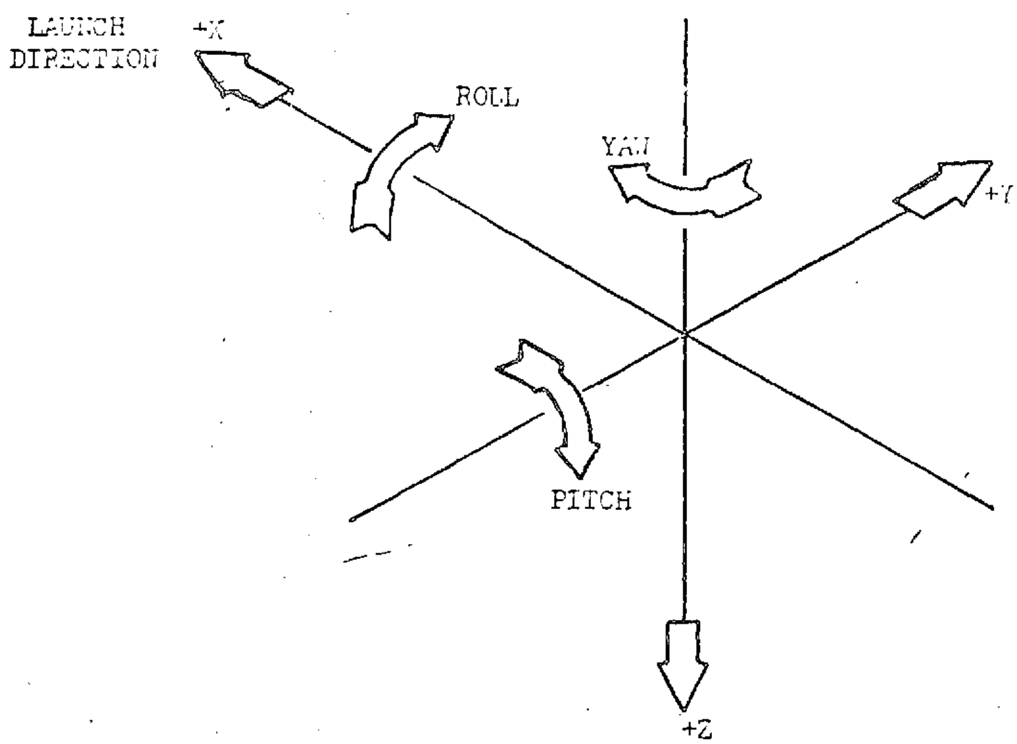

Fig. 1.1-1 Spacecraft Coordinate System

2. Titan Candidate Launch Vehicles for EOS Missions at WTR, Martin Marietta Document LV-122-73, dated October 1973. Telecon with Martin Marietta personnel revised some data contained in the referenced document.

3. Space Shuttle System Payload Accommodations, Lyndon B. Johnson Space Center Document. JSC 07700, Volume DIV, Revision B, dated 21 December 1973.

4. Performance Specification for Spacecraft Subsystems, Goddard Space Flight Center, Document EOS-410-04, dated 14 September 1973.

1.1.1.4 Basic Spacecraft

1.1.1.4.1 The GAC Basic Spacecraft described by Figure 1.1.1-2 consists of a fully insulated 58 inch long triangular shaped core structure formed by 3 vertical shear webs, and upper and lower bulkheads, and extending from the webs are six vertical trussed panels which provides the support for three 48x48x18 inch subsystem equipment modules. Each module is supported at three points as shown in Figure 1.1.1-3. In this arrangement, primary structural loads are not induced in the S/S equipment modules but are carried from the launch adapter hard points through the six rigid vertical trussed panels to the instrument support structure. This arrangement makes the subsystem modules readily removable for in flight or ground resupply at no significant design or cost impact. A S/C that initially is built to provide for fixed mounted S/S modules could easily be converted to a Shuttle resupply configuration with the addition of the resupply mechanisms described in Paragraph 1.1.1.7 again with insignificant cost and weight impact. An orbit adjust/RCS module is attached to the lower bulkhead and the mission peculiar instrument support structure mounts to the upper bulkhead of the spacecraft.

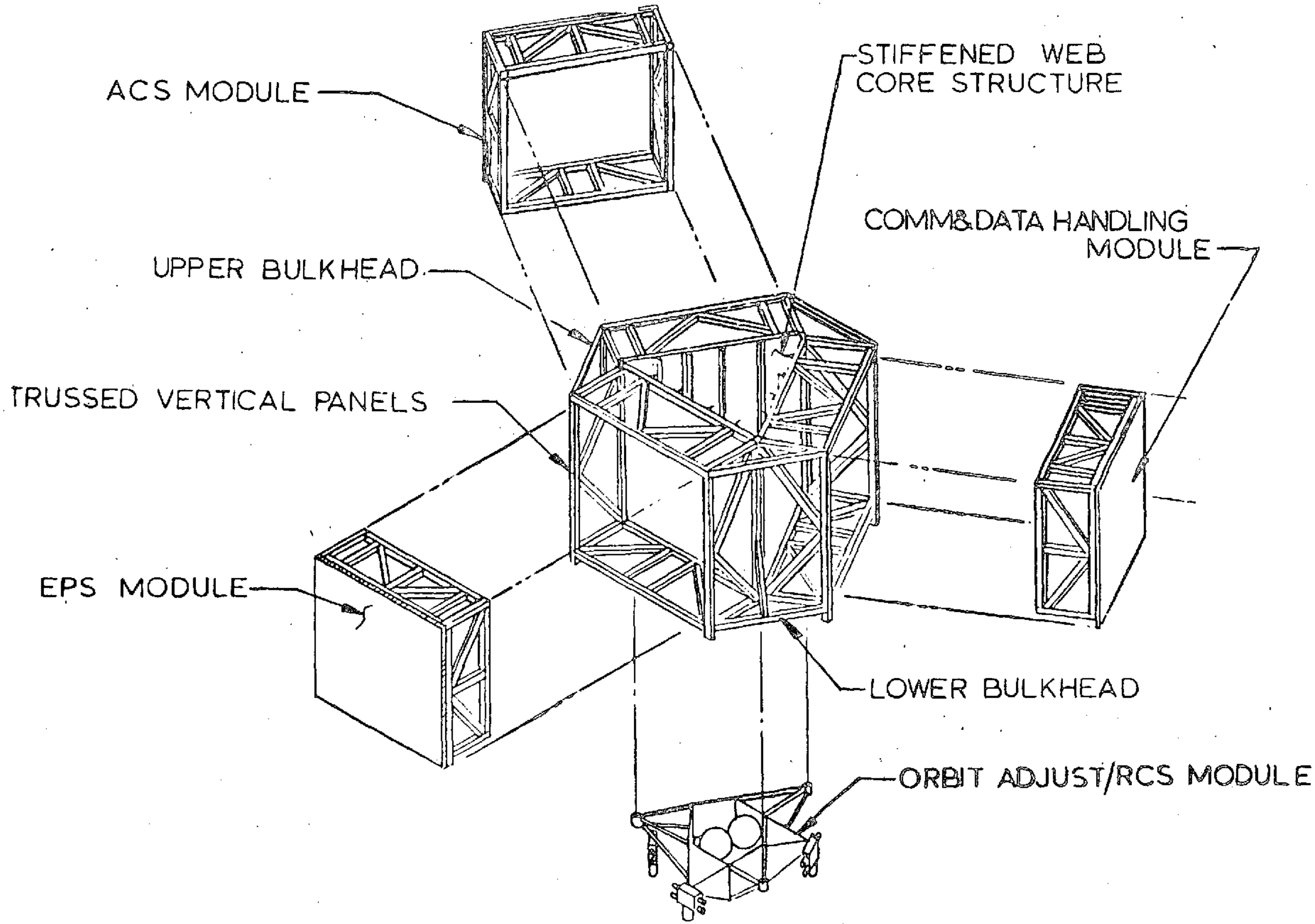

Fig. 1.1.1-2 Grumman Delta Basic Spacecraft

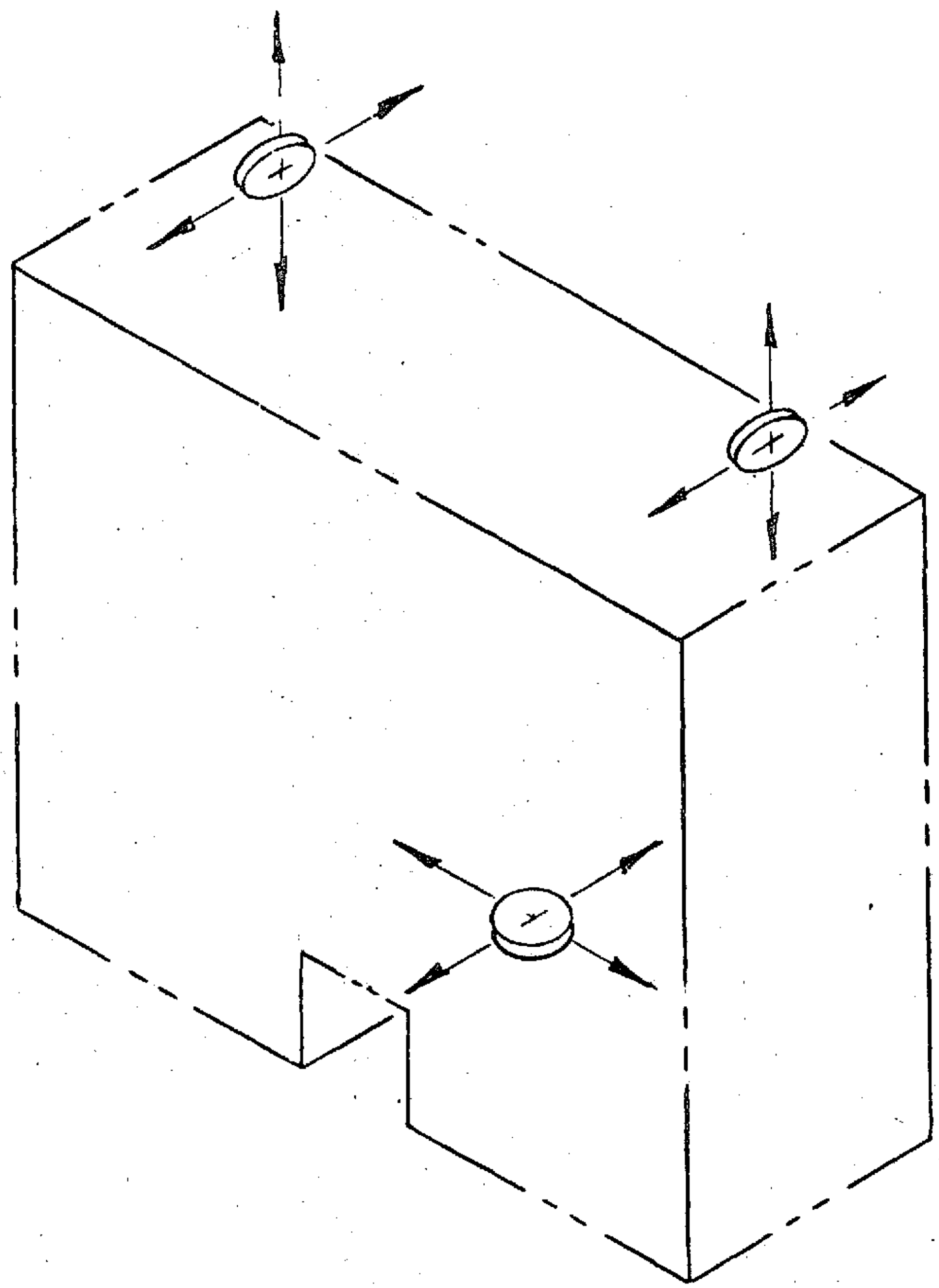

Fig. 1.1.1-3 Load Diagram Subsystem Equipment Module

1.1.1.4.2 Selected S/C/LV Adapter

This S/C configuration fits within the allowable 86 inch diameter payload envelope of the Delta launch vehicle and is mounted on the launch vehicle by means of a conventional type adapter. The adapter shown in Figure 1.1.1-4 is a conical structure approximately 24 inches high, 85 inches in diameter at the S/C lower bulkhead interface and 60 inches in diameter at the booster interface. Six (6) pyro technically actuated bolts attach the S/C to the adapter. Separation of this interface is accomplished by means of six (6) push off springs mounted in the adapter with mating pads located on the S/C structure.

1.1.1.4.2.1 Adapter Trade Study

The bottom mount adapter was selected for several reasons, the most significant of which is ease of separation of the S/C from the launch vehicle adapter. The separation is performed in an unobstructed volume with an inexpensive, lightweight, proven separation system. Table 1.1.1-6 summarizes the results of an investigation into S/C mounting methods, base mounting at this lower S/C bulkhead vs transition ring mounting at the upper bulkhead level. It concludes that the base mounted structure is stiffer due to a more direct load path and that it is lighter overall than the transition ring mount.

The GSFC Baseline Adapter which was described in reference documents as occupying the space between the launch vehicle payload fairing and the subsystem modules and their support structure posed three design problems. First, the accommodation of the triangular arrangement of three 48x48x18 inch subsystem module requires an 85 inch diameter cylindrical envelope. This configuration must fit within the maximum 86 inch diameter specified as the maximum allowable S/C static diameter in the 96 inch Delta Fairing. Clearly this ½ inch clearance all around is insufficient to contain an adequate adapter from the L/V to the S/C forward bulkhead. Second, if a wraparound adapter was made possible by decreasing the width and/or thickness of the modules, the more than 59 inch withdrawal distance renders extraction difficult. Lengthy and complex guide rails, rollers and/or

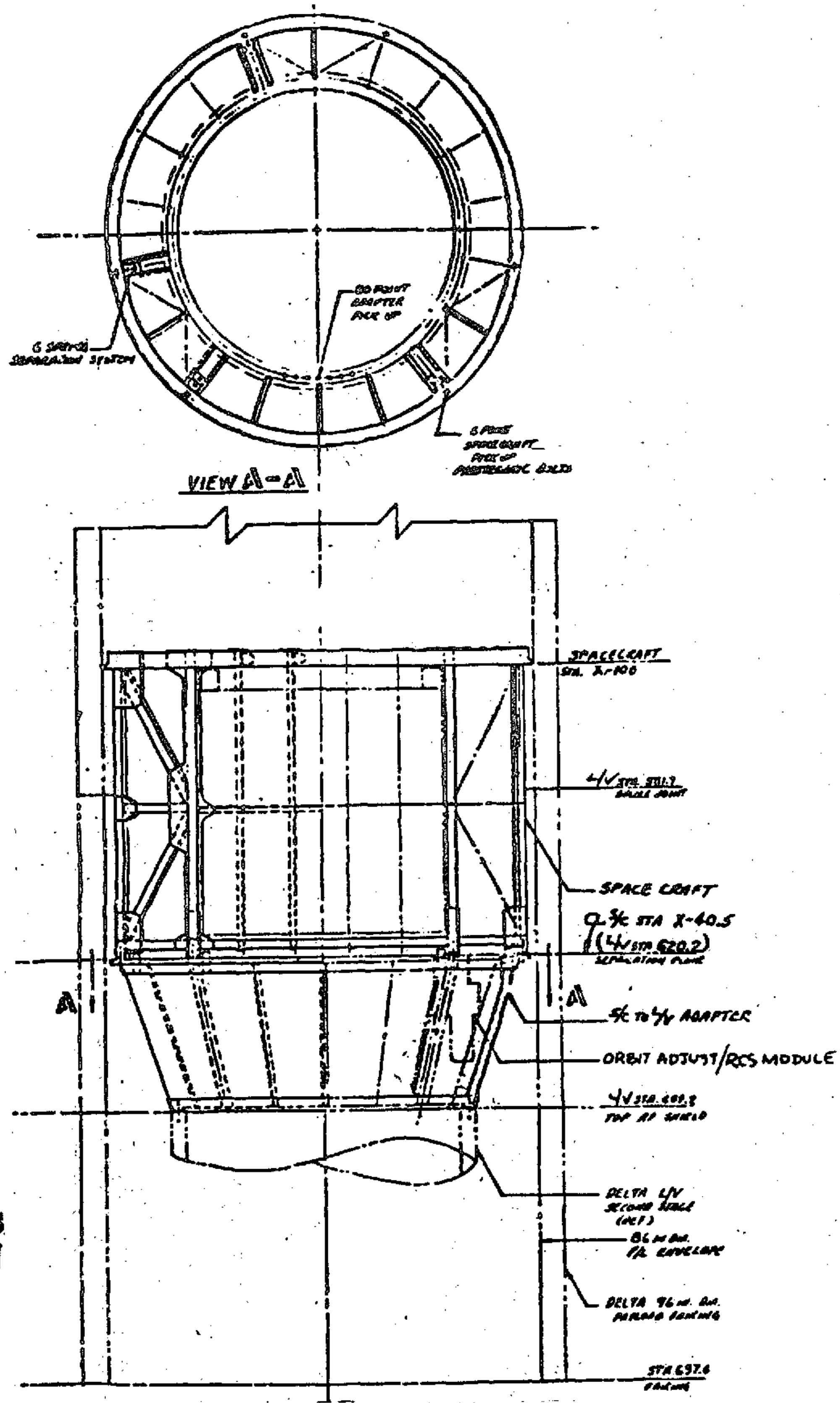

Fig. 1.1.1-4 Spacecraft/Launch Vehicle Adapter

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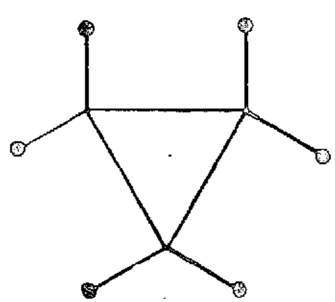
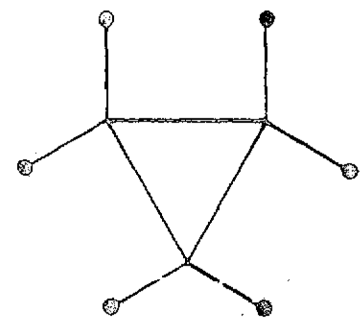
Table 1.1.1-6 Trade Study Spacecraft/Launch Vehicle Mounting

PROBLEM: For delta preferred, is there a weight difference between base mount & transition ring mount for S/C structure?





PAYLOAD SUPPORT CONFIGURATION	BASE MOUNT	TRANSITION RING MOUNT
1. PL supported on six hard points always	Most direct load path, loads carried in vertical truss members which splice into adapter longerons	Direct load path but adapter may require local reinforcement if transition ring is not stiff enough to redistribute load
2. P.L. supported on vertical beams forming the triangle or transition ring P.L. support	Load Path: Load carried into shear web by stiffener, transferred by shear of web to vertical truss and then into adapter longeron	(a) load path is the same if inst support only on vert beam. (b) if P. L. supported on transition ring not at hard points, then transition ring must be stiff enough to redistribute load without locally overloading adapter/fairing. (c) If case (b) above is always true, i.e. P.L.s always supported at any point on the transition ring, then ring must be designed for all worse case loadings. While S/C wgt. is possibly invariant with P.L. arrangement, there will be a weight penalty for all launches or new rings required for mission peculiar P.L.s.

CONCLUSION: Based on present information the base mounted delta preferred has the more direct load path system and, therefore, is stiffer. The transition ring mount system may be heavier.

Additional studies are required.

skid ways would be mandatory. Reliance on such systems could result in a moderate weight penalty. Third, an Orbit Adjust System attached below the subsystem module support structure is required which will add a minimum of 24 inches more spacecraft structure to be extracted from this rather "tight" container. Figure 1.1.1-5 illustrates the Baseline Concept and the design solutions considered by Grumman as described in the ensuing text.

Three solutions to the problem were considered and are listed below in decreasing order of complexity and weight. First, a structural modification of the Delta 96 inch payload fairing could perform the adapter function, picking up the S/C at the extremeties of the Spacecraft upper bulkhead. Such modification is heavy, costly and requires expensive fairing redesign and requalification. Such a system circumvents the Delta design load path thru the second stage, requires fairing jettison methodology and timing changes, and presupposes that the first stage exterior structure can adapt economically to a modified load pattern.

Second, varying the previous approach a separate conical stiffened sheet metal structure is attached to the L/V outer shell in the vicinity of the fairing separation plane, Delta Station 697.4, and to a separation system on the lower rather than the upper Spacecraft bulkhead similar to the Titan proposal concept. However, this design was only briefly considered because of its weight, its length, and consequently its potential lateral flexibility, and because it too ignores the designated load path through the structure provided at the upper end of the second stage.

The third and selected configuration was chosen because it picks up the existing bolt pattern at the top of the second stage guidance compartment as specified for existing adapters, i.e., the Delta 5414 attach fitting assembly. Its geometry is an inverted cone within which the Orbit adjust module fits and can be easily extracted. It utilized the shortest load path, is the lightest, 95 lbs., and does not require any structural or operational changes in the standard

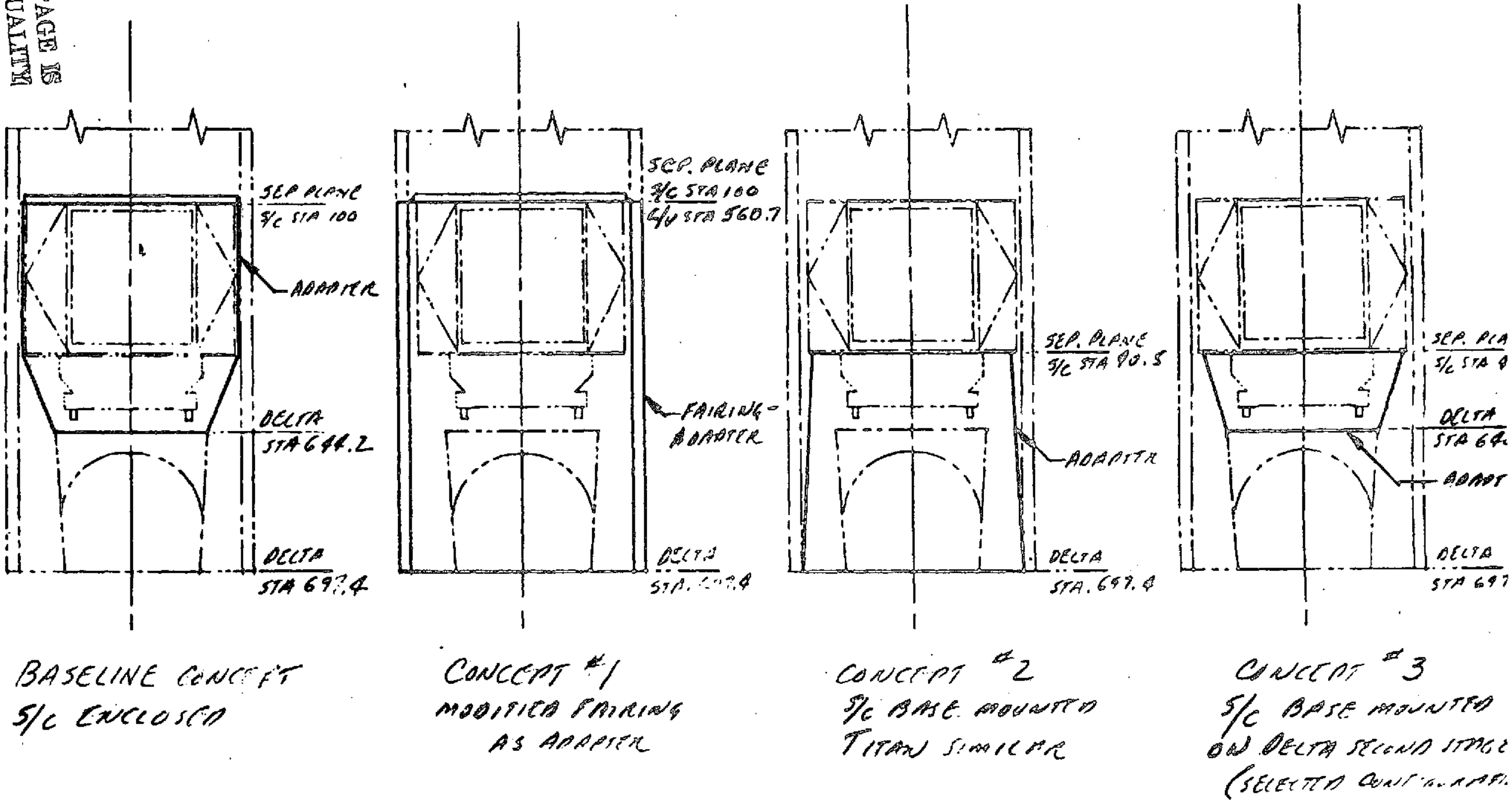


Fig. 1.1.1-5 Spacecraft/Launch Vehicle - Mounting

96 inch diameter Delta payload fairing. The configuration shown in Figure D.1.1-4 describes a 24 inch high sheet metal inverted cone capped top and bottom by shaped, extruded or machined ring frames and stiffened by 3/4 x 3/4 angles attached at 8 inch intervals to the interior surface of the cone. At the six points corresponding to the spacecraft mounting fittings the angle stiffeners are replaced by 2 x $1\frac{1}{2}$ inch channels which also house push off spring assemblies. These channel assemblies also stiffen the upper cap ring fitting locally and aid in transitioning the six concentrated loads at the top of the adapter assembly to the uniformly distributed loading in the bottom ring fitting.

The upper ring fitting is designed for containable pyrotechnic bolt or nut tie down and separation. The upper ring fitting can be alternately designed for a retainable pyrotechnically released clamp arrangement engaging the six spacecraft mounting fittings. In either case, the lower ring fitting interfaces the Delta L/V at an existing bolt circle at the upper end of the second stage.

1.1.1.4.2.2 TITAN Application of Selected Bottom Mount

When launched by a Titan III booster the spacecraft adapter retains its conic stiffened sheet character but grows considerably in length and diameter and encloses the spacecraft and the Orbit Adjust/Orbit Transfer Module. Due to the length increase, horizontal ring stiffeners at a 7 inch spacing augment the vertical angle stiffeners in stabilizing the sheet panelling. The forward ring fitting is designed as the lower section of a clamp type separation system at the spacecraft upper bulkhead. At this height the Titan Payload Fairing internal clearance envelope is approximately 109 inches in diameter and the adapter and separation system can approach this as a limiting dimension. Since the triangular module arrangement does not excede 85 inches in diameter, and the OA/OT module is no more than 60 inches in diameter no interference with the adapter is anticipated during spacecraft separation. The aft ring fitting interfaces with a 114.0 inch diameter bolt circle on an existing adapter support structure proposed by the launch vehicle contractor as shown in Figure 1.1.1-6. Note that the payload fairing is virtually unaffected by modification of the existing adapter support structure which consists of a new adapter support ring and integration of eighteen new identical adapter support fittings in the 10 inches immediately forward of the standard Titan interface at station 220.15. A minor change, the addition of an internal circumferential ring stiffener on the lower half of the Super Zip separation fitting, is the single modification to the existing P-123 Payload Fairing System normally supplied by LMSC.

This adapter concept is identical to the GSFC baseline system except that the reduced diameter of the triangular spacecraft configuration makes the upper bulkhead ring mounting practical. The square, four module spacecraft configuration is difficult if not impossible to contain within a separate S/C to L/V adapter without violating the fairing internal clearance envelope. Certainly the cost of modifying and requalifying the payload fairing to perform the adaptation would be considerable. However, in the event this approach was permissible, the added weight and cost of guides, rails, rollers, and/or skids required to insure spacecraft withdrawal makes the method highly unattractive. No weight or cost penalties have as yet been estimated but they are thought to be significant.

1.1.1.4.2.3 Spacecraft/Shuttle Mounting

The Space Shuttle launch (and retrieval) of the EOS also requires a modified clamp type separation mechanism at the spacecraft upper bulkhead. This support configuration is compatible with the Flight Service System (FSS) suggested by both the shuttle contractor and the SPAR/DSMA designers of the Special Purpose Manipulator System. The basic difference between the GAC concept and the GSFC baseline transition ring assembly is support of the six discrete mount fittings of the triangular spacecraft configuration rather than on the continuous ring system.

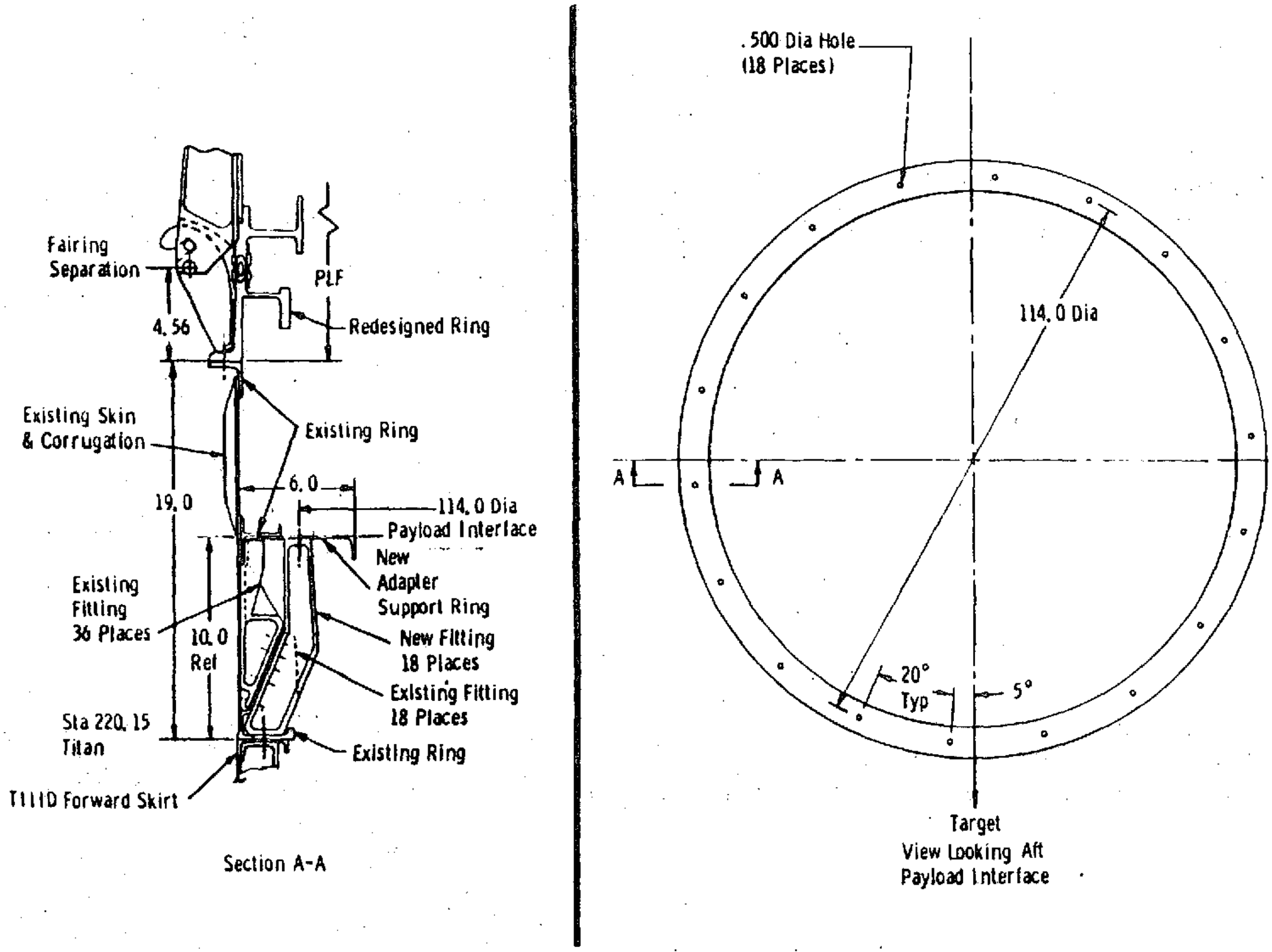


FIGURE 1.1.1-6 TITAN/SPACECRAFT ADAPTER INTERFACE

Elimination of the continuous mounting ring results in a spacecraft weight savings of 75 lbs as well as a simplification of the Flight Support System.

1.1.1.4.3 Payload Fairings

The 86 inch diameter dynamic envelope in the cylindrical section and the 6 inch radial clearance in the forward conic section of the standard 96 inch Delta Payload Fairing provides ample volume for the early EOS Instrument Payloads. The pyrotechnic fairing separation devices, being self contained, pose no shock or contamination threat to either Instrument or Subsystem components. The adapter chosen for the Delta compatible spacecraft does not in any way interface with the payload fairing. Therefore, no modification of either structure or operational function from the nominal 96 inch diameter, 1200 lb, fairing fabricated for current Delta Launch Vehicle shown in Figure 1.1.1-6A is anticipated. This payload fairing is capable of shrouding each of the EOS A, EOS A' and EOS B configurations shown in Figures 1.1.1-15,-16 and 17.

As payload sizes and weights increase beyond the Delta capability, particularly as regards the synthetic aperture radar of Mission C, the Tiros O of Mission E, and the SEOS and SEASAT-A instruments of the follow-on missions, the increased height and volume available in the 120 inch diameter LMSC P-123 Payload Fairing becomes mandatory. Figure 1.1.1-6B illustrates the fairing required for the Grumman configuration of EOS Mission C. It consits of segments A,B,C,D and G mounted on a standardized 19 inch long Payload Adapter Support Section which locates the payload adapter interface 10 inches forward of Titan Sta. 220.15. An eighteen point bolted interface is postulated based on 0.500 inch diameter holes equi-spaced on a 114 inch bolt circle previously presented in Figure 1.1.1-6 at this level.

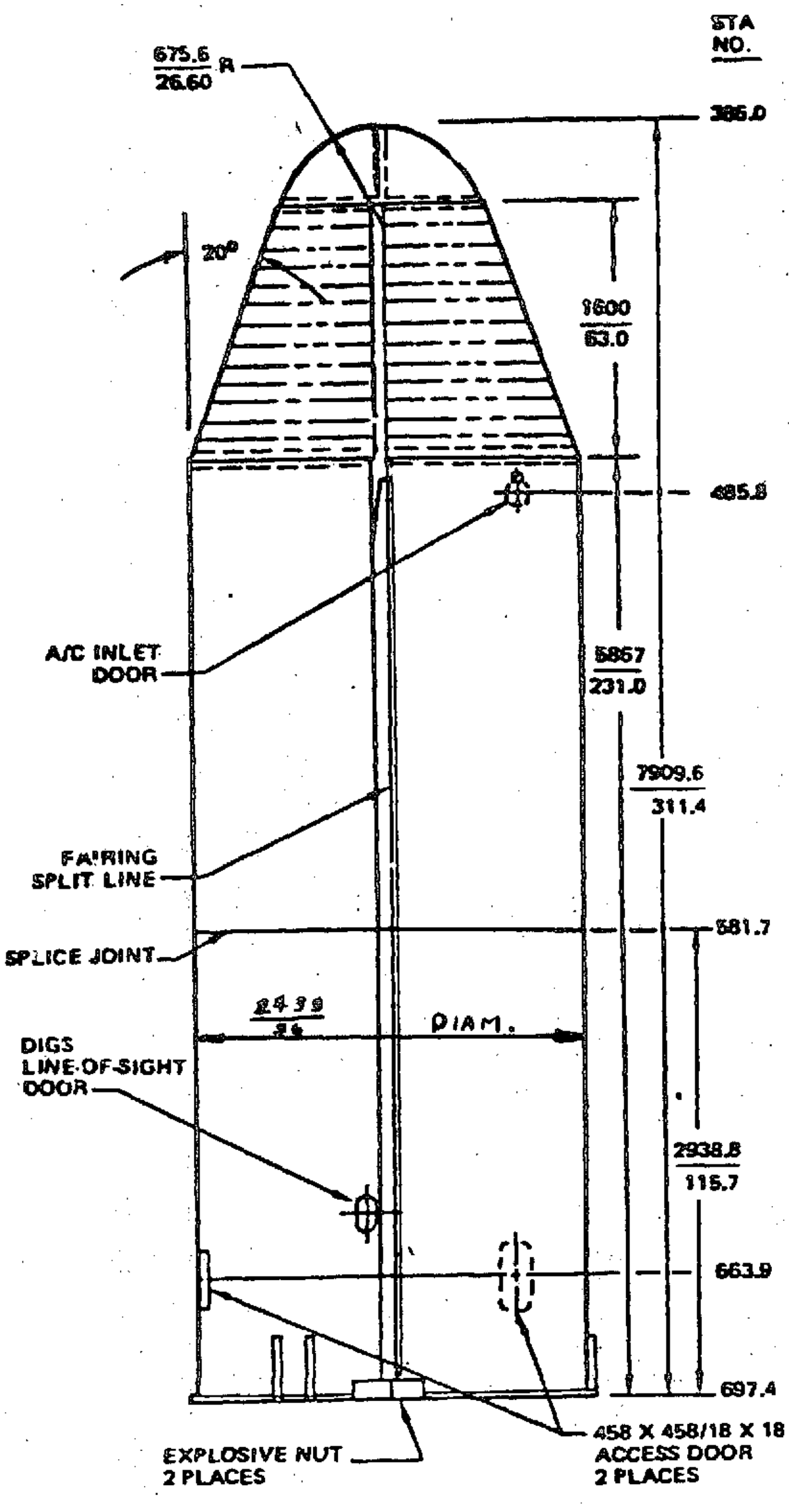


Fig. 1.1.1-6A Delta 96 Inch Fairing Profile

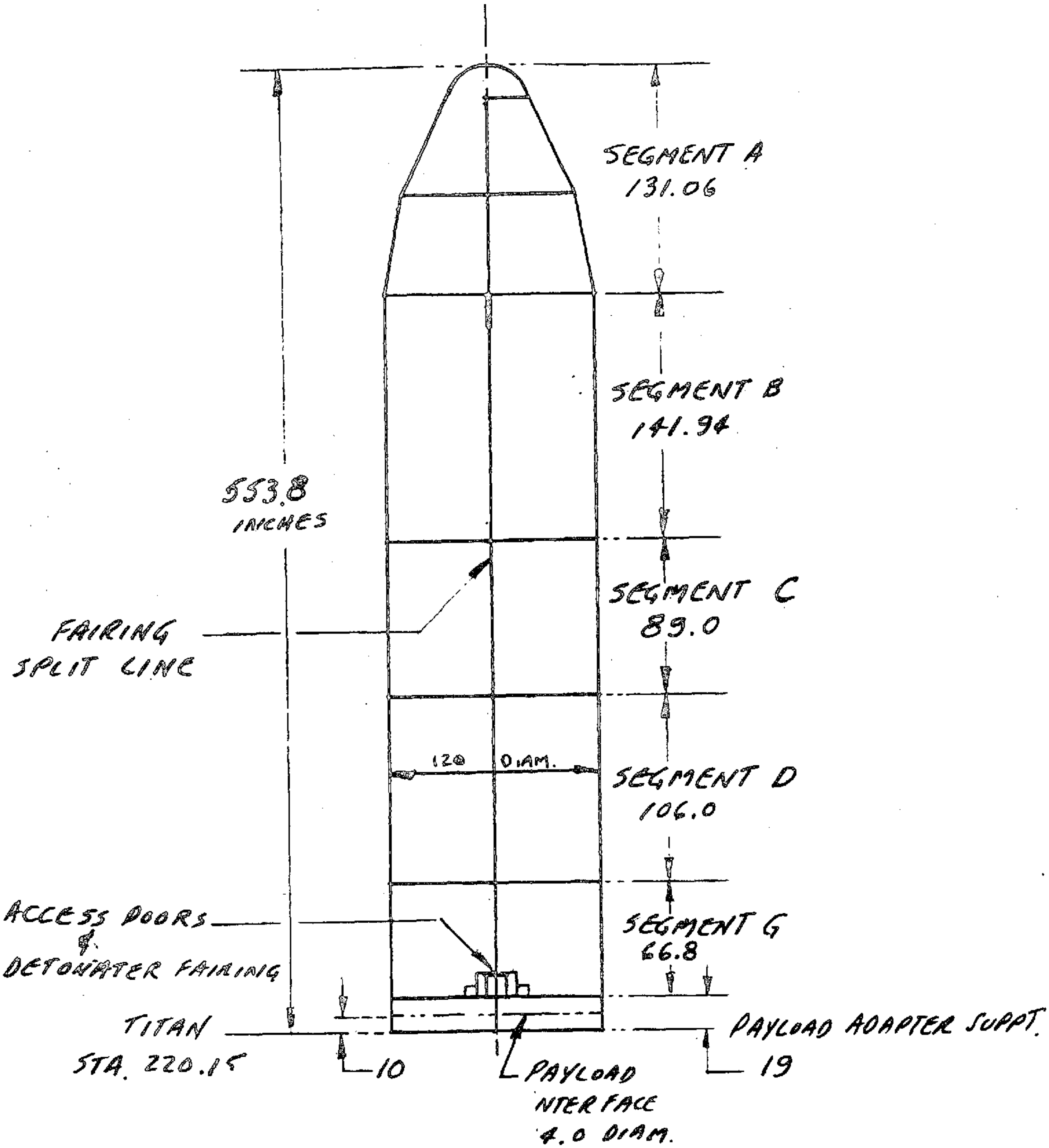


FIGURE 1.1.1-6B TITAN P.123 FAIRING PROFILE

The internal dynamic envelope of the P123 fairing is 109.2 inches diameter immediately forward of the Payload Adapter Interface. This diameter decreases at the approximate rate of .004 inches per inch of cylindrical height up to a point 409.0 inches above the interface. Above this height the envelope diameter decreases to 100.0 inches at approximately 575 inches above the payload adapter interface with a 10.0 inch radial clearance requisite in the conic section as the fairing approaches its maximum height potential of 704.5 inches overall. The separable weight of the Mission C fairing is 2492 lbs and the non-separable weight is estimated at approximately 300 lbs. The weight of the maximum height (704.5 inches) fairing is estimated at approximately 3600 lbs. total as a limiting value. The EOS Mission C as configured in Figure 1.1.1-18 is fully compatible with the previously described P-123 fairing configuration (segments A, B, C, D and G) which weighs 2800 lbs overall.

1.1.1.5 Delta S/C Core Structure

The S/C core structure is presented in Figure 1.1.1-7. The primary structure consists of three 52 x 58 inch long stiffened sheet metal shear panels mechanically joined along their sides to form a rigid triangular cross section. Gussetted, trussed bulkheads at the instrument support structure and adapter interfaces, plus 6 vertical 17 x 58 inch trussed panels between the core and the S/C mounting pickups, complete the structural assembly. The weight of this structure is estimated at 186 lbs.

The upper trussed bulkhead is framed with 1½ square aluminum alloy extruded tubing gussetted at the joints. The frame is joined to 3 corner posts and six outriggered vertical panels at the external shuttle/spacecraft support points. Its internal members attach to and form the cap members for the three core shear webs. The outer diagonally braced 20 x 42 inch square tubed panels provide the upper, primarily vertical, support for the S/S modules. This arrangement provides 3 inboard and 6 outboard hardpoints for instrument section structure base support.

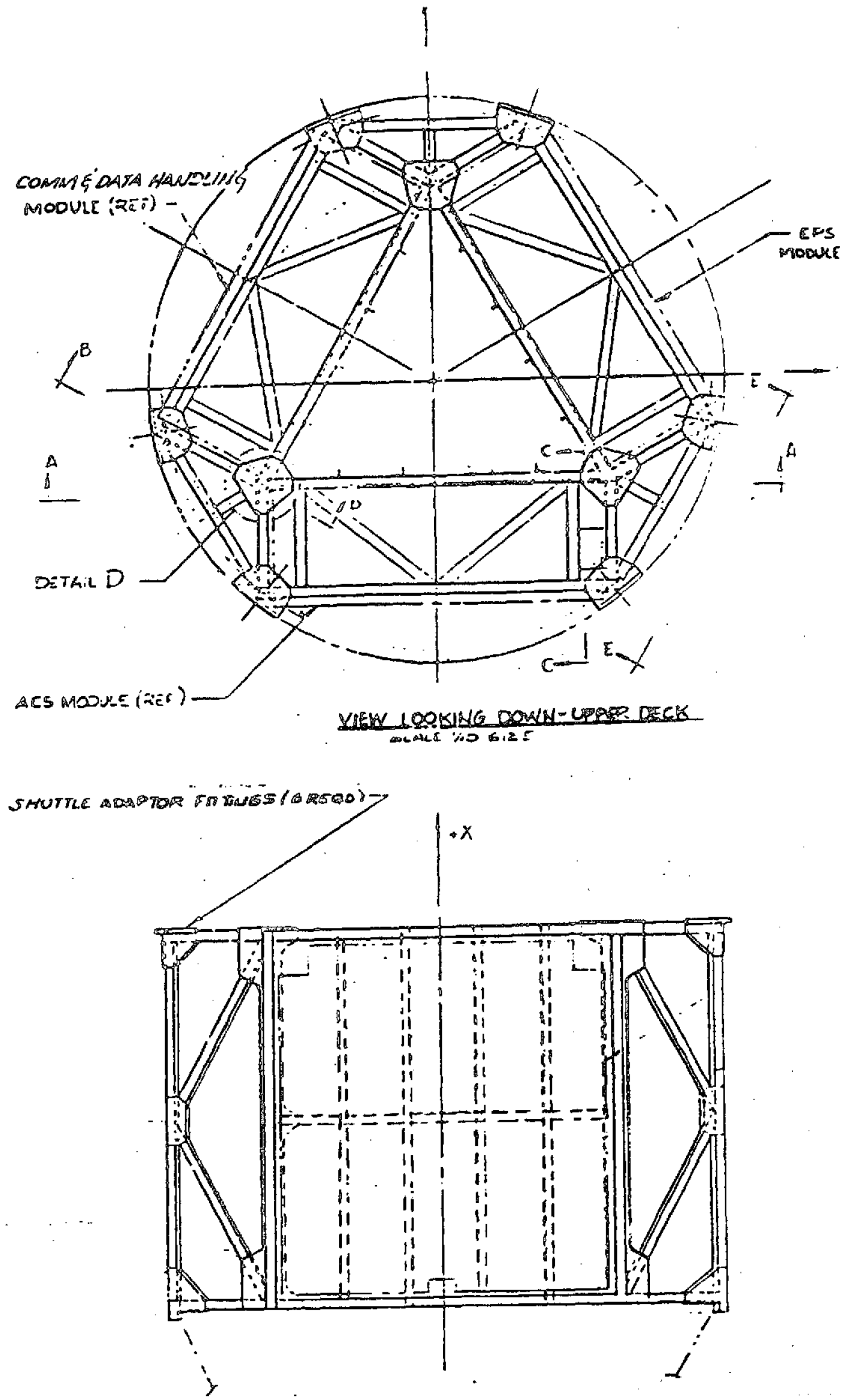


Fig. 1.1.1-7 Delta Basic Spacecraft Structural Arrangement

In addition, the deep sections between the three interior posts are capable of beaming intermediate instrument loads laterally to the vertically stiff end posts. When mated with the core webs, lower bulkhead and the six vertical panels, the frame has the capability to either unload vertically to the Shuttle interface mechanism of the Flight Support System (FSS) at this level or to transfer instrument and subsystem module loads to a six point interface with a Delta Launch vehicle adapter, at the aft bulkhead 58 inches below.

The 52 x 58 inch core web end fittings form the 3 corner vertical posts. Each of the three webs are stiffened vertically by extruded angles and a single horizontal member. The webs provide for the redistribution of vertical, horizontal and torsional shear loads between the inboard vertical posts. This configuration is effective in providing maximum multi-directional lateral stiffness for minimum weight.

The six cross-braced vertical outrigger panels are also constructed of extruded tubing 17 x 58 inches, one side of each panel is an inboard corner post which also acts as the core web edge member. When launched by a Delta vehicle the outriggers accept loads from both the instrument section support structure and the subsystem modules as redistributed by the upper and lower truss bulkheads and core webs. These loads are subsequently delivered as vertical and horizontal column and/or tension loads at the six discrete mounting fittings at the lower bulkhead/Delta adapter interface. The outboard vertical posts are connected in pairs by horizontal diagonals for increased stability. Each of the diagonals also acts as a support for one of three docking probes used for S/C retrieval.

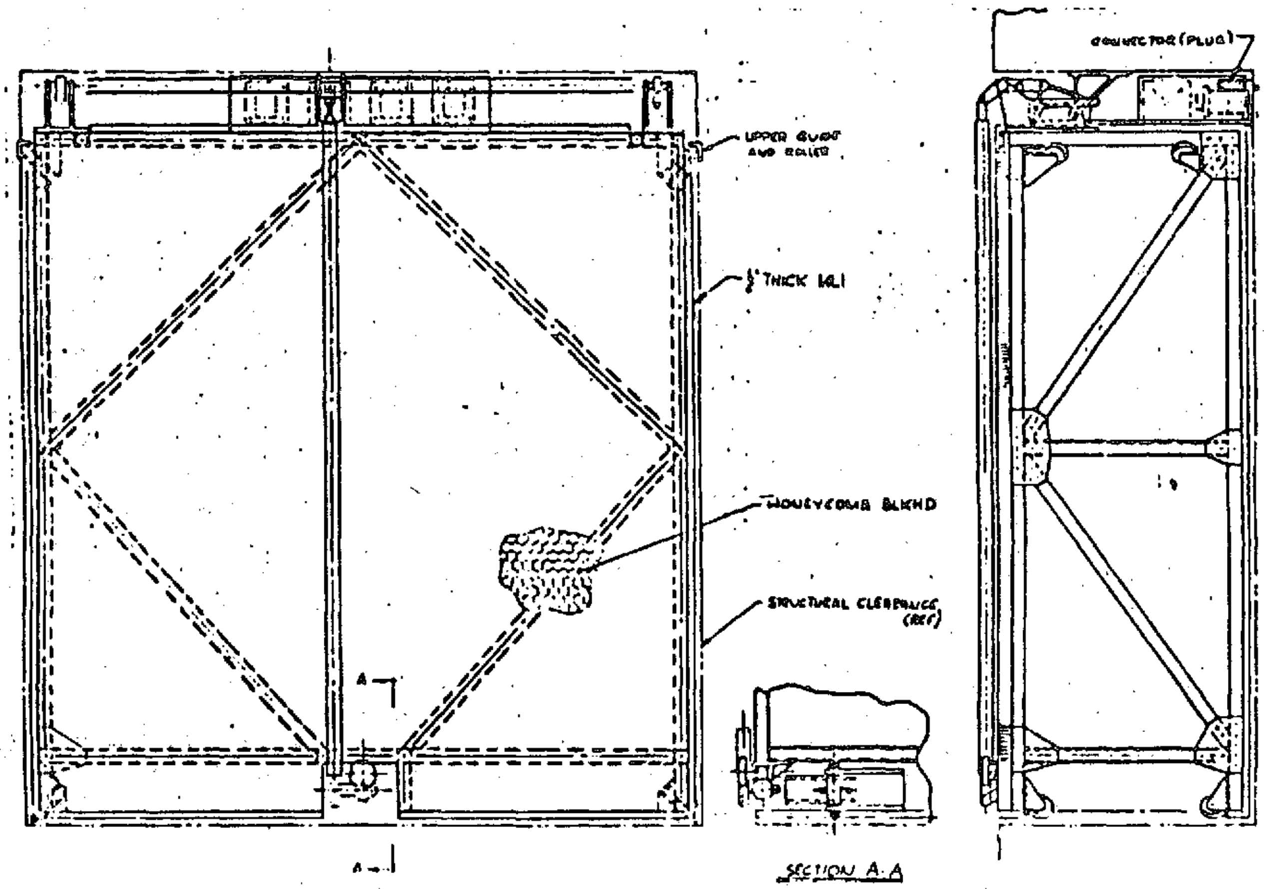
The lower trussed bulkhead is constructed of square extruded tubing, connects with the core shear web corner posts, acts as the lower cap of the shear webs and accepts and redistributes loads similarly as its upper counterpart with the exception that the lower S/S module latch design permits essentially only lateral load transfer to its supporting structure rather than the basically vertical load distribution of the module upper latches.

The most significant design driver for the structure is the stiffness required to meet the launch vehicle design frequencies in both the lateral and longitudinal direction. The stiffened sheet metal core structure meets this requirement and with this structural arrangement primary structural loads are not induced in the S/S equipment modules.

1.1.1.6 Subsystem Module Structure

The Grumman structural design for a typical subsystem module features structural and thermal isolation and introduces a unique 3 unit latching mechanism operable from a single control point as illustrated in Figure 1.1.1-8. The structural elements of the 48 x 48 x 18 inch module and the mechanical details of the latching and guide system are shown in this structural arrangement drawing.

The module is framed out conventionally with square tubing arranged in roughly 18 x 24 inch cross-braced panels. The design is compatible for either riveted gusset or welded joint construction. Square intercostal tubes are installed locally in the module side frames where required for latch or connector installation. Subsystem components can be similarly provided for, although their primary mounting surface is the interior surface of the 48 x 48 x 1 inch thick aluminum honeycomb panel which is the module outer surface. Eight guide and roller assemblies to insure module alignment are shown in this configuration. This number could be sharply reduced without seriously affecting the alignment function as follows; removal of the two rollers at the bottom outboard corners; and removal of all four upper rollers and guides and relocating one roller guide assembly vertically at the center of the top side panel near the outer face of the module. In this generalization the interior framework is stabilized by tubular cross-braces in a diamond pattern. This structure is almost certain to be replaced by the vertical and horizontal thru partitioning required by specialized subsystem components and/or groupings.



SUBSYSTEM MODULE
STRUCTURAL ASSEMBLY

FIGURE 1.1.1-8

The local structural stiffening for the three point over center latching mechanisms naturally utilizes extruded square tubing identical to the framing members. The locations of these stiffeners and/or the lower cutout in the honeycomb panel are identical module to module unless displaced by or combined with adjacent component unique reinforcement. Similarly the guide racks and rollers are standardized and will not be redesigned or relocated except as necessitated during customizing of the design to ACS, Comm and D/H, or EPS function. It is anticipated that provisions for the attachment of aluminized mylar insulation, particularly on the side panels, can be standardized. However, radically differing requirements will preclude this design possibility on the outer, radiating surface of the honeycomb panel.

Thermal isolation of the individual modules is enhanced by the minimal conductive paths implicit in rotating mechanical latches onto 3 individual pins. Further thermal isolation is assured by use of non-metallic rollers and/or alignment guides. The thermal characteristics of the outer honeycomb surface will be controlled by coatings and devices tailored to the requirements imposed by the particular subsystem installed in the module.

Despite the alternative core centered load path suggested by Grumman the basic module structure differs only slightly from the given baseline in respect to its truss framework and honeycomb outer panel. The basic difference lies in the 3 point latching mechanism and guide/roller system.

Structural provisions for the 3 unit coordinated latching system and the standard extrusions and/or components anticipated in the roller/guide system are obviously fewer and less complex than the 4 unit individually operated tracked mechanisms. Additionally, the elimination of one-fourth of the mechanisms increases overall system reliability significantly while reducing overall module structural detail and assembly cost.

Because launch loads are constrained entirely to the maximally stiffened core structure, subsystem components whose alignment is critical need not be mounted on heavily loaded and therefore flexible module structures. Similarly, lightly loaded modules are unlikely to experience or cause permanent structural deformations under launch loads making tracked replacement difficult if not impossible. This inherent alignment stability plus the positive over center latching assures that alignments once set during ground calibration will be maintained and repeatable in orbit.

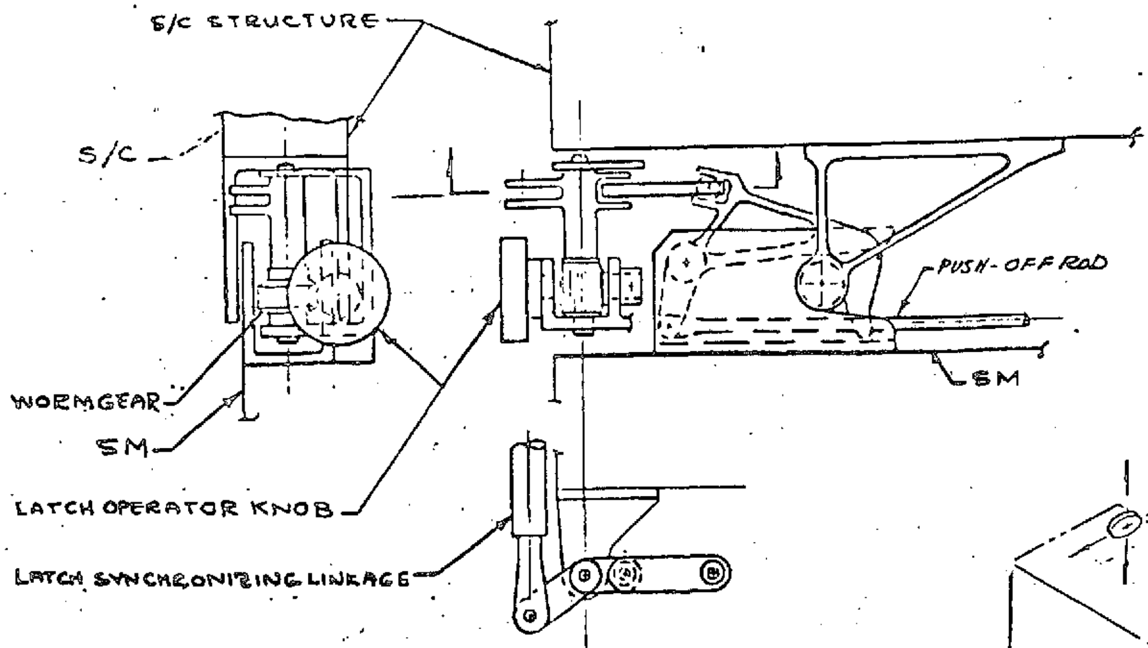
Finally the ease of totally interrupting thermal conductive paths between the module and adjacent support structures is evident. At a number of points between the module structure and its supports, non-metallic or low conductivity materials may be used for pins, bushings, bracketry, and rollers thereby permitting thermal isolation using multi layer insulation blankets.

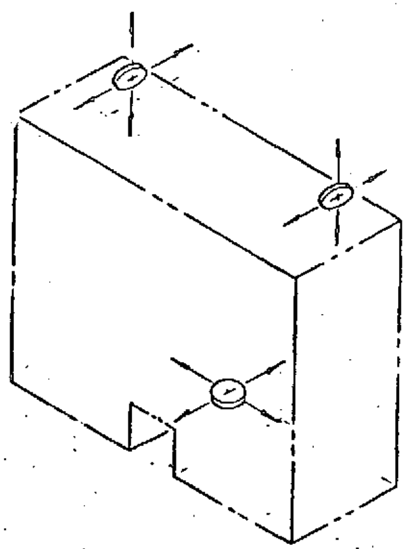
The Weight of the Baseline 'Dry' module is 98 lbs plus 68 lbs for latch mechanisms for a total of 166 lbs. For three modules the weight differential GSFC to GAC designs is 498 vs 249 lbs or 249 lbs total in favor of the Grumman design.

1.1.1.7 Resupply Approach

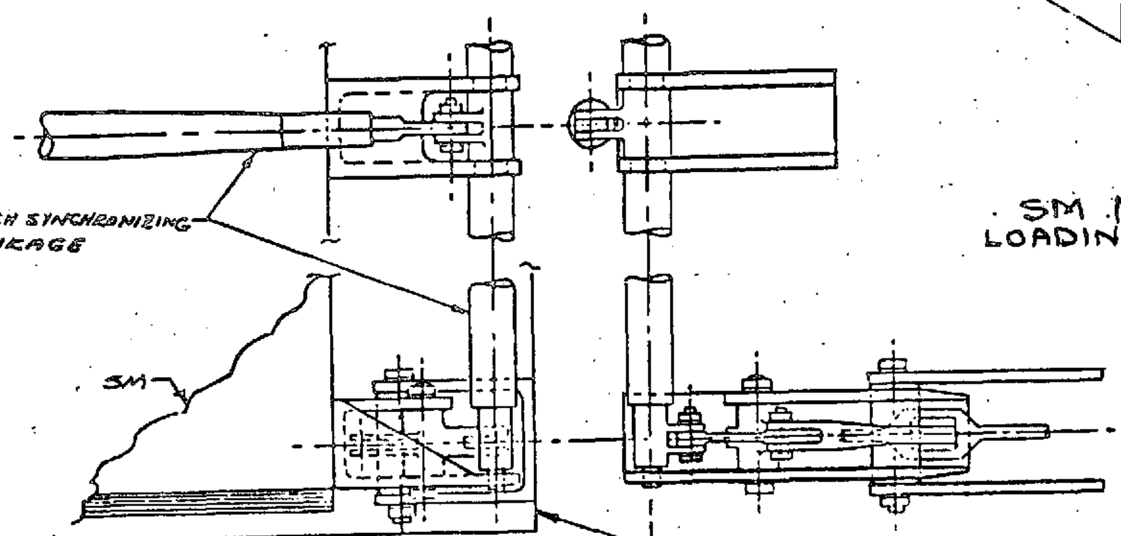
1.1.1.7.1 Subsystem Modules

The GAC mechanism shown in Figure 1.1.1-9 consists of three hook and roller latches utilizing a self-locking linkage. Initial alignment is accomplished by means of a track and roller guide system and final alignment is achieved by means of the latch roller guides provided on each of the three latches. The latch hooks are configured to supply the final pull-in force required for mating of the self-aligning electrical connector and the latch operated push-off rods supply the necessary demating force. Launch loads are carried via the 3 latch points only and no appreciable S/C loads are transmitted through the track and roller





SM MODULE
LOADING DIAGRAM



LATCHING MECHANISM
SUBSYSTEM MODULES
TYPICAL CORNER LATCH

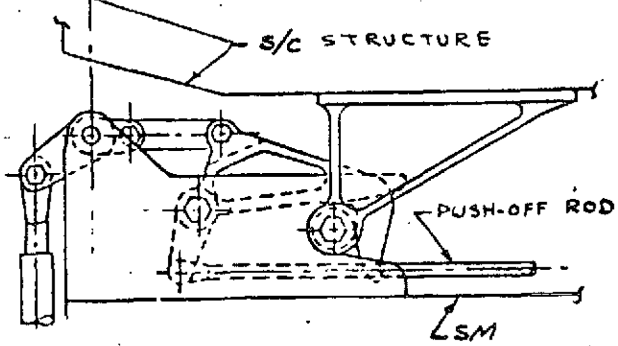


Fig. 1.1.1-9 Latching Mechanism Subsystem Modules - Typical Central Latch

guide system. SM positioning and latch operation is accomplished by means of a single latch operator and is readily adaptable to a dual or triple latch operator arrangement. The latch operator consists of a holding knob riggedly fixed to the SM and containing a centrally located rotary driver which supplies rotary input to the worm gear set operating the latches. A common latch operator is utilized for all the resupply latches.

The GSFC Baseline Subsystem Modules latch mechanism consists of four screw-jacks and four guide rails, and carries launch loads across the guide rails via a preload device. The GSFC Delta Baseline does not provide resupply capability. See Table 1.1.1-7, -8, and -9 for rationale and summary.

1.1.1.7.2 Instrument Assemblies, Orbit Adjust Stage, Solar Array Configurations

The selected GAC baseline utilizes the identical (SM) mechanization concept for the resupply of the Instrument Assemblies as shown in Figure 1.1.1-9. The resupply latch system for the Orbit Adjust stage is similar to the SM system and differs only in that one of the three latches is replaced by a conical socket engagement shown in Figure 1.1.1-10. The Solar/Array/Drive resupply latching is similarly accomplished with dual SM type latches and a third point support provided by a conical socket engagement shown in Figure 1.1.1-11. Additional retract latches are provided on the Solar Array for the purpose of sustaining loads during launch, orbit adjust and shuttle re-entry.

The GSFC Baseline does not specifically define resupply latching systems for the Instrument Assemblies, Orbit Adjust Stage or Solar Array.

The GAC Delta Baseline is configured to be compatible with a resupply capability similar to that of the GAC Titan configuration. Therefore resupply latching provisions can be omitted on an option basis.

1.1.1.7.3 Solar Array Accommodations

1.1.1.7.3.1 Flexible Solar Array

The flexible Solar Array shown in Figure 1.1.1-11 is of the roll-up flexible type mounted on a deployable support frame. The stowage and deployment mechanism

Table 1.1.1-7 Rational for Rating Latches

CRITERIA	HOOK & ROLLER (GAC)	SCREWJACK (GSFC)
Weight	10 lbs/Module	68 lbs/Module
Cost	o Less weight o Less complexity	o Extremely tight manufacturing Tolerances required
Positioning Repeatability	A 3 pt. engagement is inherently more conducive to position accuracy o Low System friction minimizes structural distortions o 3 pt. System can be made to accommodate thermal distortion	o Greater component accuracy required for positioning o High latching friction o System cannot accommodate thermal gradients
Reliability	o 3 latches are more reliable than 4 o All rotating joints can be made redundant o No sliding contact o No grease lubrication	o Redundancy difficult to achieve for sliding contact o Reliable grease lubrication in space is difficult to maintain.
Ease of in Orbit Operation	o Synchronization problem is greatly minimized o Low latching loads reduce in orbit difficulties	o 4 screw jacks are difficult to synchronize o Grease lubrication can result in contamination of other systems o High latching loads add additional complexity to in orbit resupply

TABLE 1.1.1-8

RESUPPLY CAPABILITY

	TITAN BASELINE		DELTA BASELINE	
	GSFC	GAC	GSFC	GAC
SUBSYSTEM MODULES	Yes	Yes	No	Yes
INSTRUMENT ASSY'S	No	Yes	No	Yes
ORBIT ADJUST STAGE	No	Yes	No	Yes
SOLAR ARRAY	No	Yes	No	Yes

TABLE 1.1.1-9

COMPARISON OF LATCH SYSTEMS

SCREWJACK vs HOOK & ROLLER

	HOOK & ROLLER (G A C)		SCREWJACK (G S F C)	
TRADE CRITERIA	RATING	FACTOR	RATING	FACTOR
o WEIGHT	Excellent	(4	Non-Optimal	(0
o COST	Good	(2	Fair	(1
o POSITIONING REPEATABILITY	Good	(2	Fair	(1
o RELIABILITY	Good	(2	Fair	(1
o EASE OF IN ORBIT OPERATIONS	Good	(2	Fair	(1
RATING TOTALS		12		4

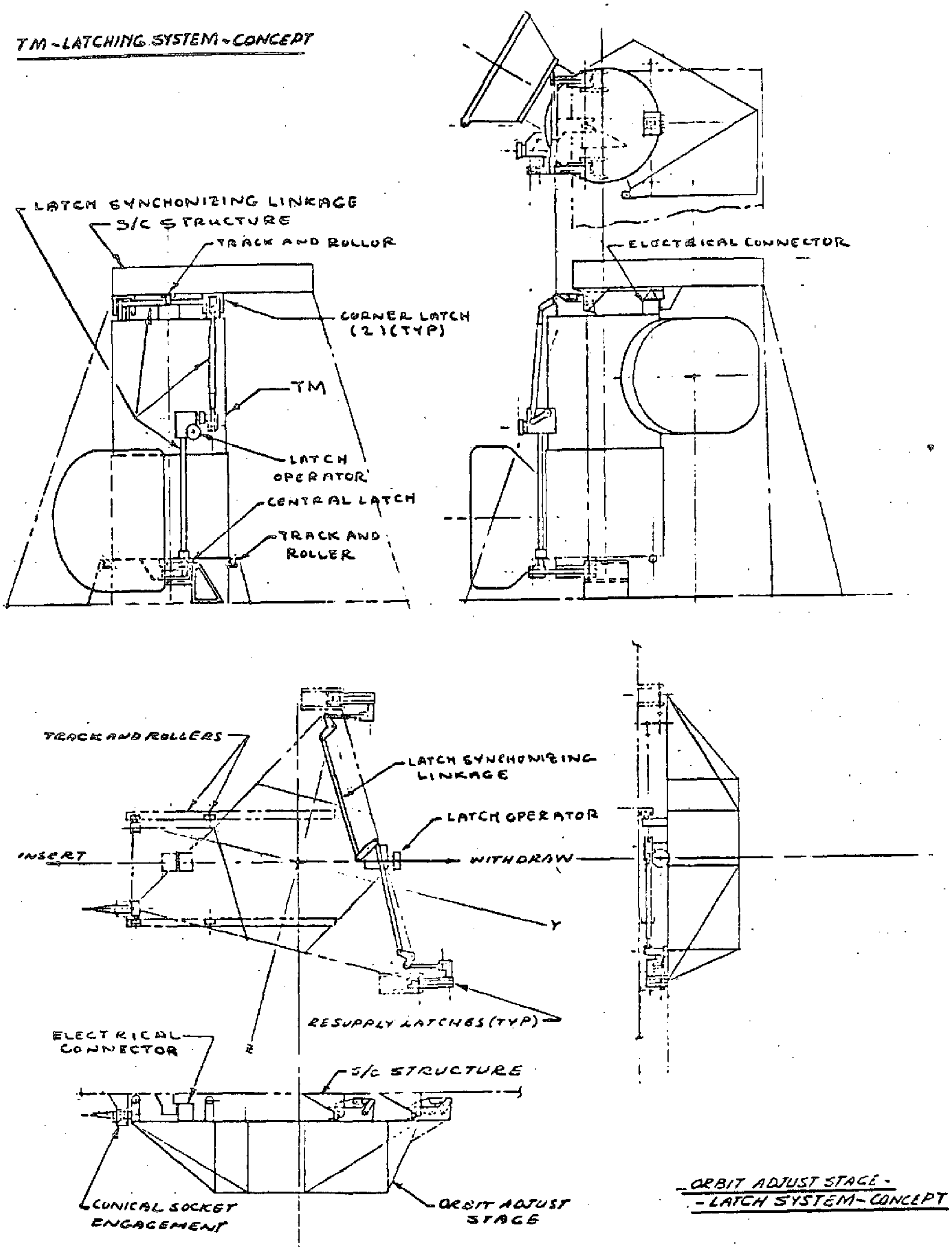


Fig. 1.1.1-10 Resupply Mechanisms for TM and OAS

FLEXIBLE SOLAR ARRAY MECHANISM

FOR STOWAGE, DEPLOYMENT, RETRACTION AND RESUPPLY

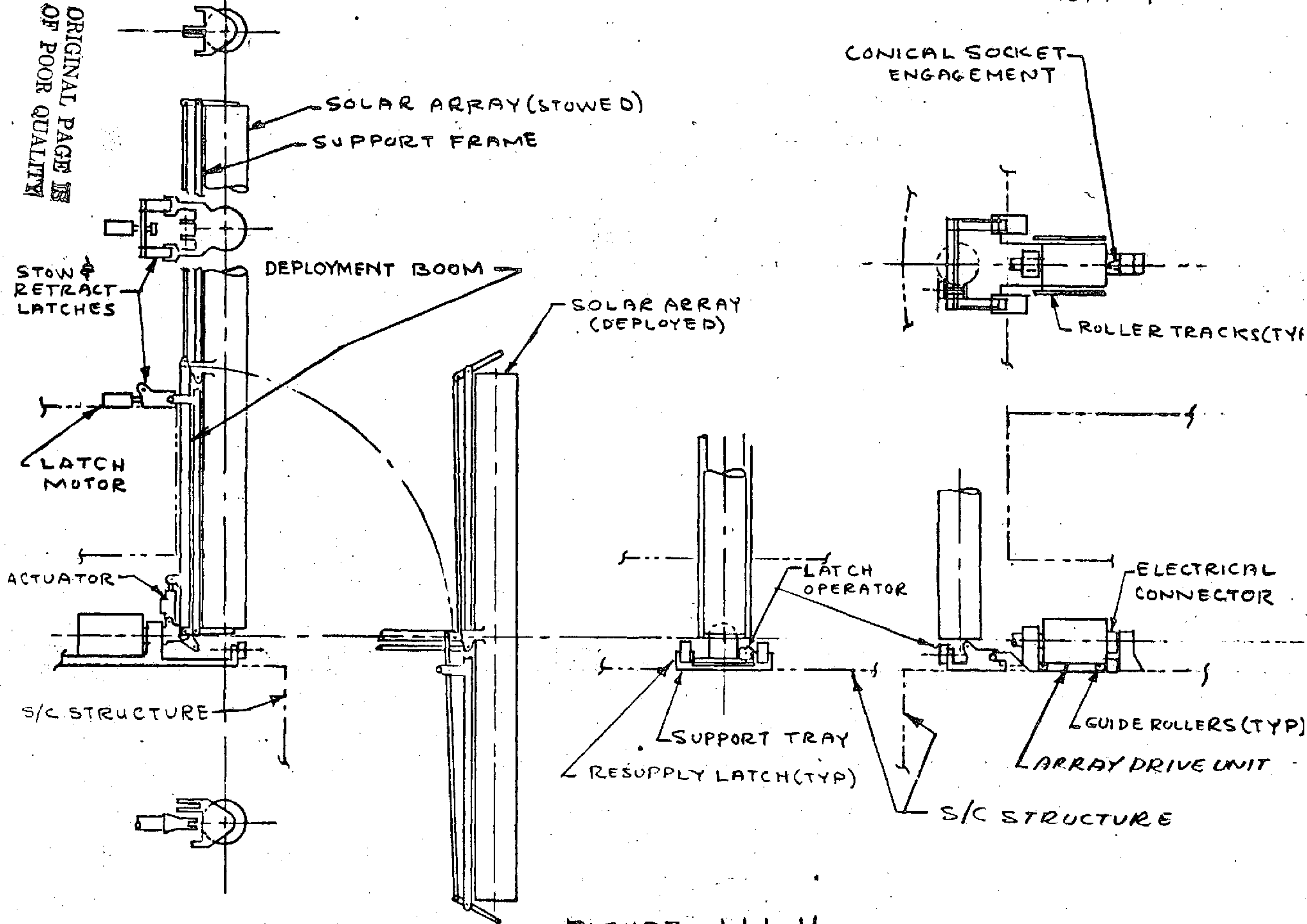


FIGURE 1.1.1-11

is configured to provide retrieval and resupply capability. A single electric screwjack actuator operates the deployment mechanism both during deployment and retraction. For stowage, dual hook and roller latches are provided to secure the Array Frame to the spacecraft structure during launch orbit adjust and shuttle re-entry. In addition, both the frame and deployment boom is snubbed against the structure in the stowed condition. The stow latches are actuated by an electric motor driven worm gear set. The resupply system consists of dual hook and roller latches and a single conical socket engagement mounted on a support tray which houses the array drive motor and the deployment boom lower support. Guide rollers are provided on the tray to facilitate initial alignment during its insertion into the spacecraft. Insertion and removal is accomplished by grasping the single latch operator knob and latching or unlatching is affected by rotation of a drive socket within the knob. Thrust forces for mating and demating the electrical connector are supplied by a hook pull-in and a push-off rod respectively. Therefore, only a torqueing force need be supplied by the SAMS end effector.

1.1.1.7.3.2 Rigid Solar Array Mechanism

The rigid solar array shown in Figure 1.1.1-12 is configured with four rigid solar panels. The two inner panels are hinged to the deployment boom and the remaining two outer panels are hinged to the inner panels. A panel actuator mounted on the deployment boom operates the two inner panels by means of a deployment linkage. The two outer panels are actuated by a linkage operated by the inner panels. A boom actuator is used to deploy the array away from the spacecraft prior to panel deployment. A series of stow latches are utilized to provide support during launch and retrieval. The resupply mechanism consists of dual hook and roller latches with a conical socket providing the third point support. The boom assembly is coupled to the array drive unit which in turn is mounted on a support tray. A single latch operator is employed to resupply the solar array and drive unit assembly.

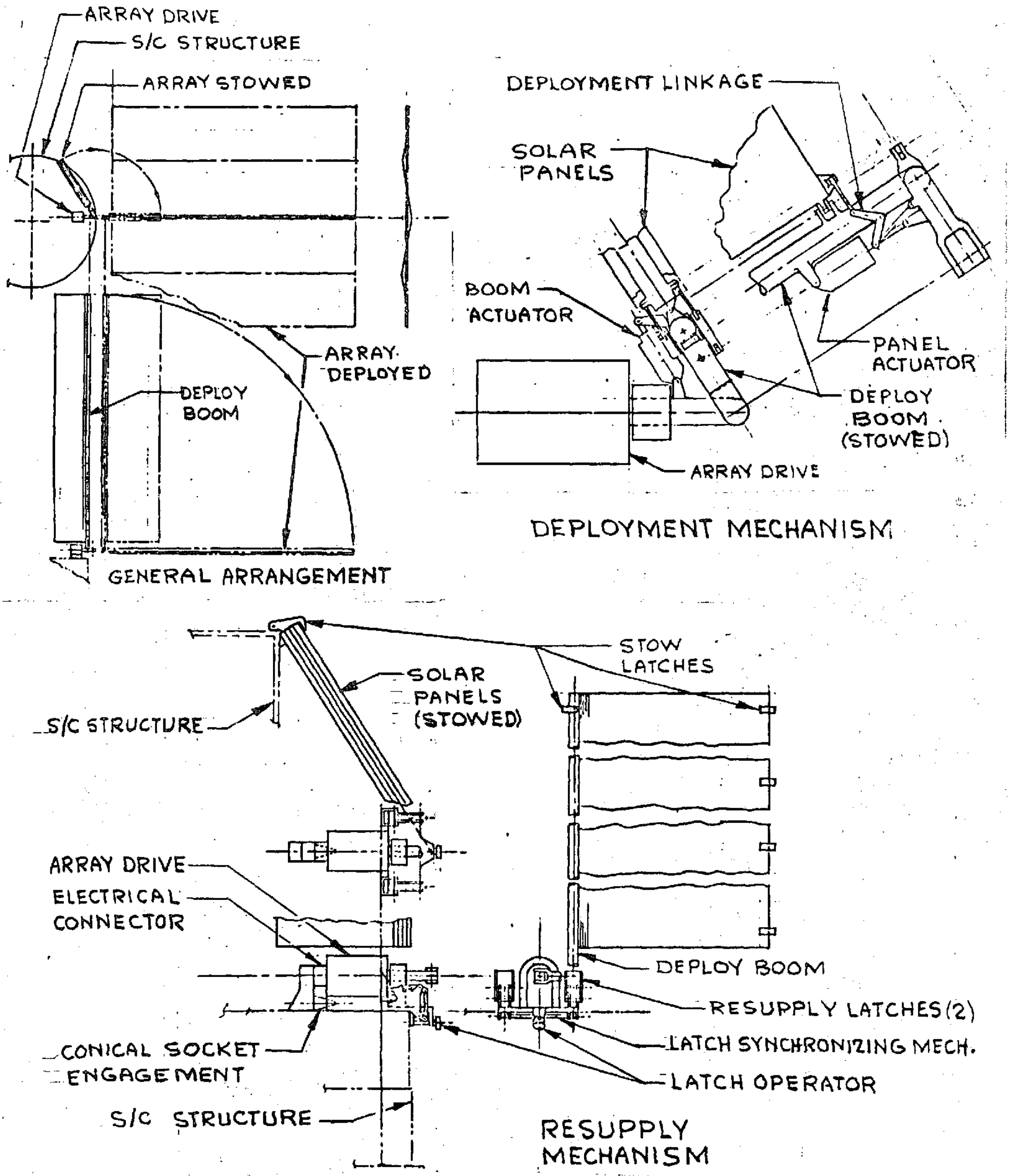


Fig. 1.1.1-12 Solar Array Configuration

1.1.1.8 Orbit Ajust

The Orbit Adjust/RCS Module shown in Figure 1.1.1-13 provides support for four Thruster Pods and two 10 inch diameter hydrazine propellant tanks. Each Pod houses two 0.1 lb., two 1.0 lb., and one 5.0 lb. thrusters.

The stage consists of a central hexogonal module which contains the propellant tanks. The module is 12 inches deep and 24 inches across the flats. Six corner tee members are connected by stiffened sheet webs. A honeycomb shelf is attached to the bottom cap angles of the peripheral webs to support the propellant tanks. Five square tubes join the upper opposite corners of the hexagon with the aid of a splice plate at their central intersection.

Four square tube struts extend off each of three alternate sides of the cental hexagon and terminate at the stage attachment/latch fittings which are 120° apart on a 30 inch radius.

Two tapered, stiffened sheet beams extend off two of the remaining three sides of the central hexagon to provide support for two of the Thruster Pods. The other two pods are supported off the underside of two of the stage support strut assemblies by sheet metal brackets and angles.

The Thruster Pods consist of a sheet metal C section which forms the back, top and bottom of the module, two removable end plates, to which are attached the low level thrusters, and an outer cover which serves as an access panel and module closure. The hi-level thruster is attached to the bottom of the C section. As most of the module structure is of relatively simple, straight line geometry, "standard" sections are used wherever possible.

1.1.1.9 Titan Spacecraft Core Structure

The additional instruments contemplated for EOS follow-on and the choice of the Titan III B launch vehicle initiated the trial design of the cruciform core structure presented in Figure 1.1.1-14. The increased diameter of the Titan payload fairing legislated a wider diameter S/C to L/V adapter and therefore

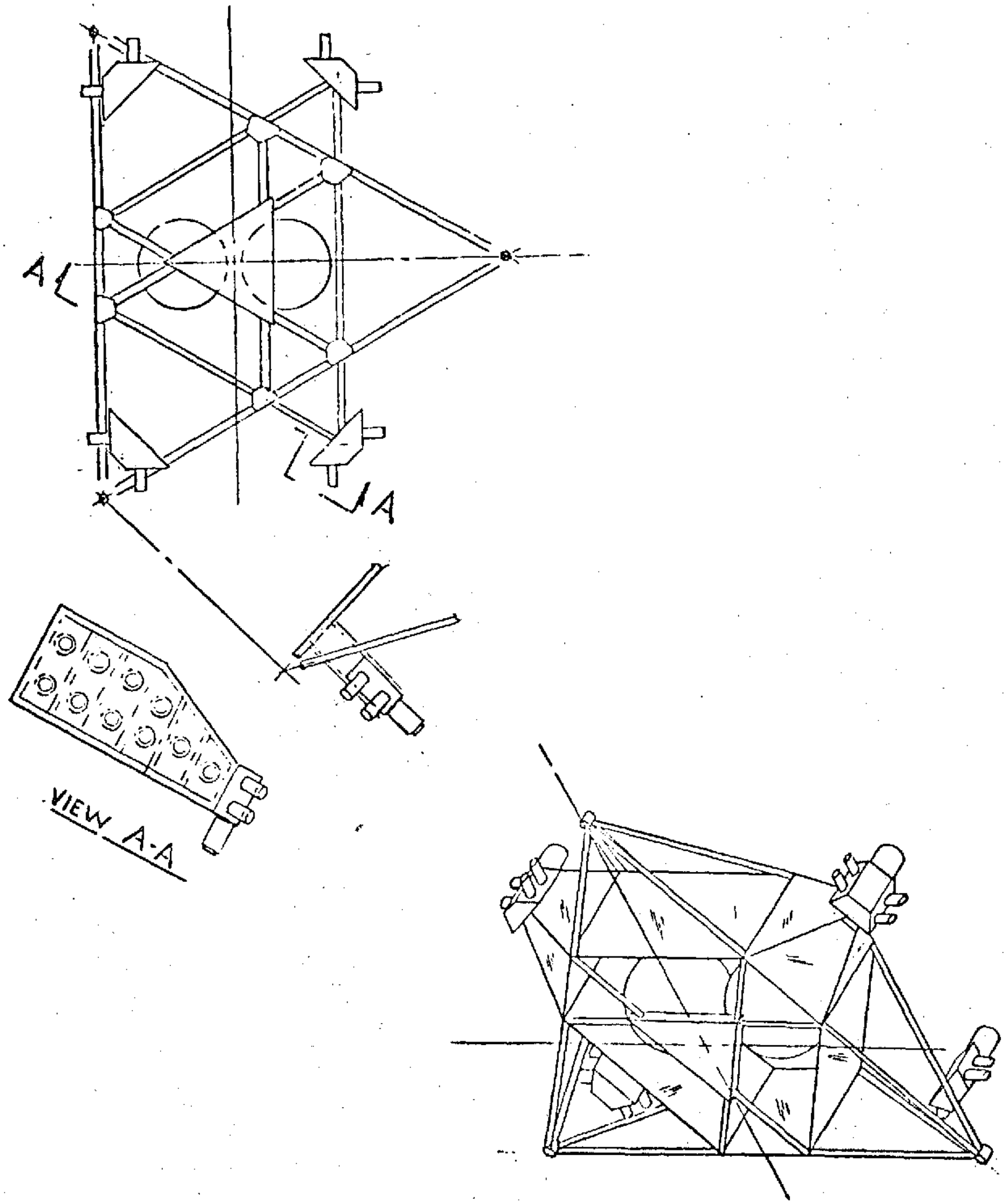


Fig. 1.1.1-13 Orbit Adjust/RCS Module

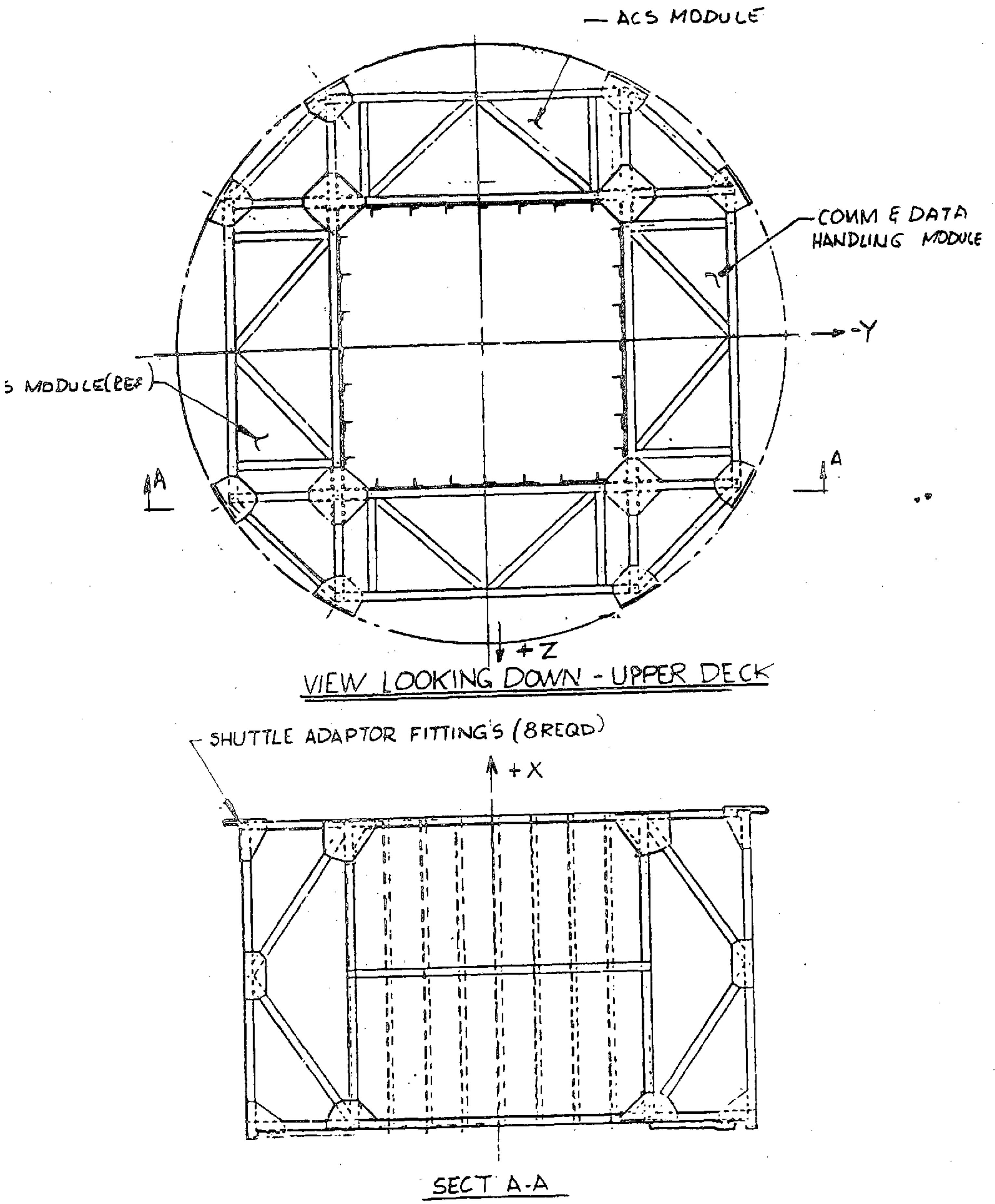


Fig. 1.1.1-14 TITAN Basic Spacecraft Structural Arrangement

a wider core structure base. Additionally, the anticipated increase of subsystem components particularly in the Data Handling subsystem indicates the possible requirement of a fourth module to accommodate the overflow. The addition of this module and the severe lateral dynamic stability design requirements immediately suggested the inherently stiff 52 inch x 52 inch x 55 inch long stiffened sheet metal shear web square core. Gussetted truss bulkheads at the instrument support structure and adapter interfaces plus eight vertical 20 inch by 55 inch truss panels between the core and the S/C mounting points complete the structure assembly. The weight of this structure is estimated at 360 lbs. and carries a recurring cost of $ K using rudimentary throw away jigs and tooling.

The forward truss bulkhead is framed out with $1\frac{1}{2}$ to 2 inch square aluminum alloy extruded tubing gussetted at the joints to be compatible with either welded or mechanical joining techniques. The frame is joined to 4 corner posts inboard and eight outrigger vertical panels at the external shuttle/spacecraft support points. Its internal members attach to and form the caps for the four stiffened sheet core shear webs. Outer diagonally braced 20 inch by 42 inch square tube panels provide the upper primarily vertical and lateral support for the subsystem modules. This arrangement provides 4 inboard and eight outboard hardpoints for instrument structure base support. In addition, the deep sections between the 4 interior posts are capable of beaming intermediate instrument loads laterally to the vertically hardened end posts. The frame is designed to accept lateral crushing loads from the proposed clam shell shuttle attach fitting. When mated with the core webs, the lower bulkhead and the outrigger vertical panels, the frame has the dual capacity to either unload vertically to the Shuttle interface mechanism at this upper level or to transfer instrument and subsystem module loads to an eight point interface with a Titan launch vehicle adapter at the aft bulkhead level 55 inches below.

The 52 x 55 inch aluminum alloy core web end fittings are the 4 inboard vertical posts. Each of the four webs are stiffened vertically by extruded angles spaced 7 inches apart and by a single horizontal member. They have the function of redistributing vertical, horizontal and torsional shear loads between the inboard vertical posts. This configuration is most effective in providing multi-directional lateral stiffness.

The eight cross-braced vertical outrigger panels are also constructed of square extruded tubing 20 inches by 55 inches, one side of which is the inboard vertical post which, in turn, is the core web edge member. They are designed to accept the loads of the lower module latches from the lower bulkhead frame and transfer them as shear loads to the core webs to be taken out ultimately at the Shuttle/spacecraft interface. When launched by a Titan vehicle the outriggers accept loads from both the instrument support structure and the subsystem modules as redistributed by the upper and lower truss bulkheads and inner core webs. These loads are subsequently delivered as vertical and horizontal column and/or tension loads at the eight discrete mounting fittings at the lower bulkhead/ Titan adapter interface. Note that the outboard vertical posts are connected in pairs by horizontal diagonals for increased stability.

The lower truss bulkhead constructed of square extruded tubing, connects with the shear web corner posts, acts as the lower cap of the core shear webs and accepts and redistributes loads similarly as its upper counterpart with the following two exceptions. Primarily, no significant crushing loads will be applied to the aft fittings because of the radial stiffness of the adapter forward ring fitting. Secondly, the lower subsystem module latch design permits essentially only lateral load transfer to its supporting structure rather than the lateral and vertical loading patterns built into the module upper latches.

Grumman is convinced that because of the severe, worst case lateral and longitudinal dynamic loading envelope applied to this structure, stiffness rather than strength is the governing design criteria and that pound for pound and dollar for dollar the sheet metal core structure is a serious competitor to the open truss I-Beam construction of the baseline. The GSFC baseline support structure as configured in the reference drawings supplied GAC is estimated at 580 lbs. The GAC alternate concept is only 360 lbs per current estimates. This 220 lb saving is significant in itself but even more so when the 249 lb (83 lb/ module x 3 modules) module weight saving is added for a total 469 lbs in "dry" structure and latch mechanisms alone.

D.1.1.10 Instrument Accommodations

D.1.1.10.1 EOS-A Instrument Section

The initial instrument payload chosen for configuration study included a 5-Band multi spectral scanner similar to the design for ERTS B, a Thematic Mapper proposed by the Hughes Aircraft, Space & Communications Division, and a Data Collection System concept originated at GSFC for EOS. Both the MSS and the TM include in their designs a thermal radiator which ideally requires a 180° field of view of "Black Space." Instrument geometry and sensor fields of view plus the essentially "morning orbit" require radiator location on the +Y side of the spacecraft, offset approximately 130° from the Nadir. The optical design of the MSS requires that it be mounted with its longitudinal (long) axis perpendicular to the S/C line of flight and the TM requires its longitudinal axis to be parallel to flight vector with the optical view port on the forward or leading (+X) end. The DCS antenna is fixed and specifies a Nadir field of view. Its electronics is part of an Instrument Mission Peculiar Module.

Figure 1.1.1-15 illustrates the payload and ancillary tape recorders, antennae and solar array arranged for maximum efficiency on the spacecraft structure. The TM is mounted parallel to the line of flight toward the shaded (-Y) side of the spacecraft to permit maximum exposure of its radiator. The solar array is mounted on center line toward the -Y side of the spacecraft to take advantage of the increased solar impingement in this area. An Instrument Mission Peculiar Electronics Module is mounted directly forward of the solar array on the Nadir (+Z) face near the C&DH module. The rectangular tape recorder module is similarly positioned immediately forward of the TM. The MSS is mounted directly above the TM on a beam/platform and projects its radiator assembly on the shaded side of the spacecraft directly above the TM radiator. The DCS antenna is attached directly to the Nadir face of the upper beam/platform. A fixed X-Band antenna, Nadir pointing, is supported by a light tubular truss structure off this beam/platform.

The TM base is mounted on a Lower Box Beam/Platform 37 in long x 26 in wide x 15 inches deep whose beam caps are fastened to and supported by the caps of two of the vertical shear webs of the spacecraft core structure. The upper mounts of the TM are attached to the underside of a trapezoidal beam/platform 37 in long x 10 in. deep with one 38 inch and one 26 inch side. This upper beam is attached to the lower beam along the forward (+Z) edge by a stiffened sheet metal bulkhead notched to clear the TM field of view and sunshade. A structural panel connects the upper and lower support along the common 26 inch beam width (-Y side). In addition, six tubular struts support the upper beam from 5 hardpoints on the spacecraft forward bulkhead adding the required stiffness to raise the natural frequency above requirements. A three point determinate support for the instrument has been assumed and ample clearances with surrounding structure has been provided to employ the Grumman latching system.

The MSS is mounted at 3 points on the upper face of the trapezoidal platform and is treated as a cantilevered load. Simple diagonally braced tubular trusses

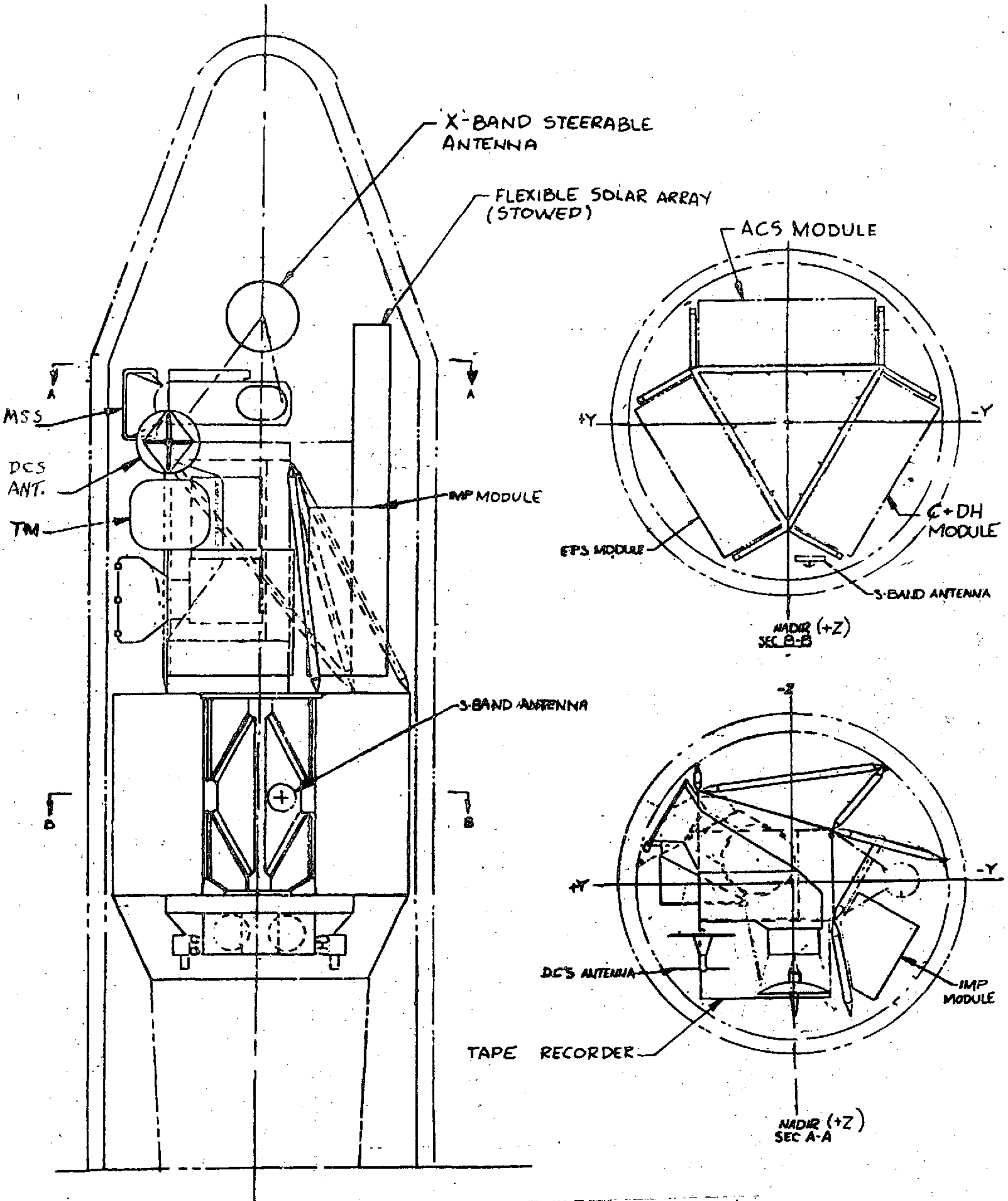


Fig. 1.1.1-15 EOS-A Configuration Delta L/V

are used to support the DCS and X-Band antennae. The flexible Solar Array is supported by an additional beam or a continuation of the Lower Beam/Platform attached to the spacecraft forward bulkhead capable of longitudinal $\pm$X loading. Lateral loads in the Y and Z direction are resisted by reaction of the bulkhead and by two struts extended from the upper trapezoidal beam/platform.

The Tape Recorder Module and Instrument Mission Peculiar Module are latched to the vertical bulkhead walls which connect the upper and lower beam/platforms and to the upper spacecraft bulkhead using the Grumman 3 Point Latching System.

During resupply operations, see Figure 1.1.1-15A, the TM is manipulated and removed parallel to the axis in the +Y direction. The MSS may be extracted from the (-Z) side while the tape recorder module is removable from the opposite or Nadir (+Z) face. The Instrument Peculiar Module is removed and replaced along an axis displaced 45° from both the Y and Z axes on the Nadir side of the spacecraft.

1.1.1.10.2 EOS A' INSTRUMENT SECTION

The viewing requirements of the HRPI and MSS result in the configuration shown in Figure 1.1.1-16. Both instruments view the Nadir while the MSS has the additional requirement of a radiator viewing "Black Space."

The basic support structure for both instruments is a "5 Sided Box" where the lower and upper faces are Box Beams and the other three sides are Strut-Trusses. The HRPI is positioned between the lower and upper faces and the MSS is on top (+X) of the upper face. The lower Box Beam (-X) reacts directly into two of the three shear webs of the subsystem structure below. In addition, three struts from the spacecraft "hardpoints" to the upper Box Beam add stability in the Y, Z planes. The assumed three point support and resupply capability of these instruments require a special latch/retension system, and clearances to the structure have been allowed for them. HRPI removal for resupply is in the -Z direction, the MSS is in the +Z direction. The solar array is supported on

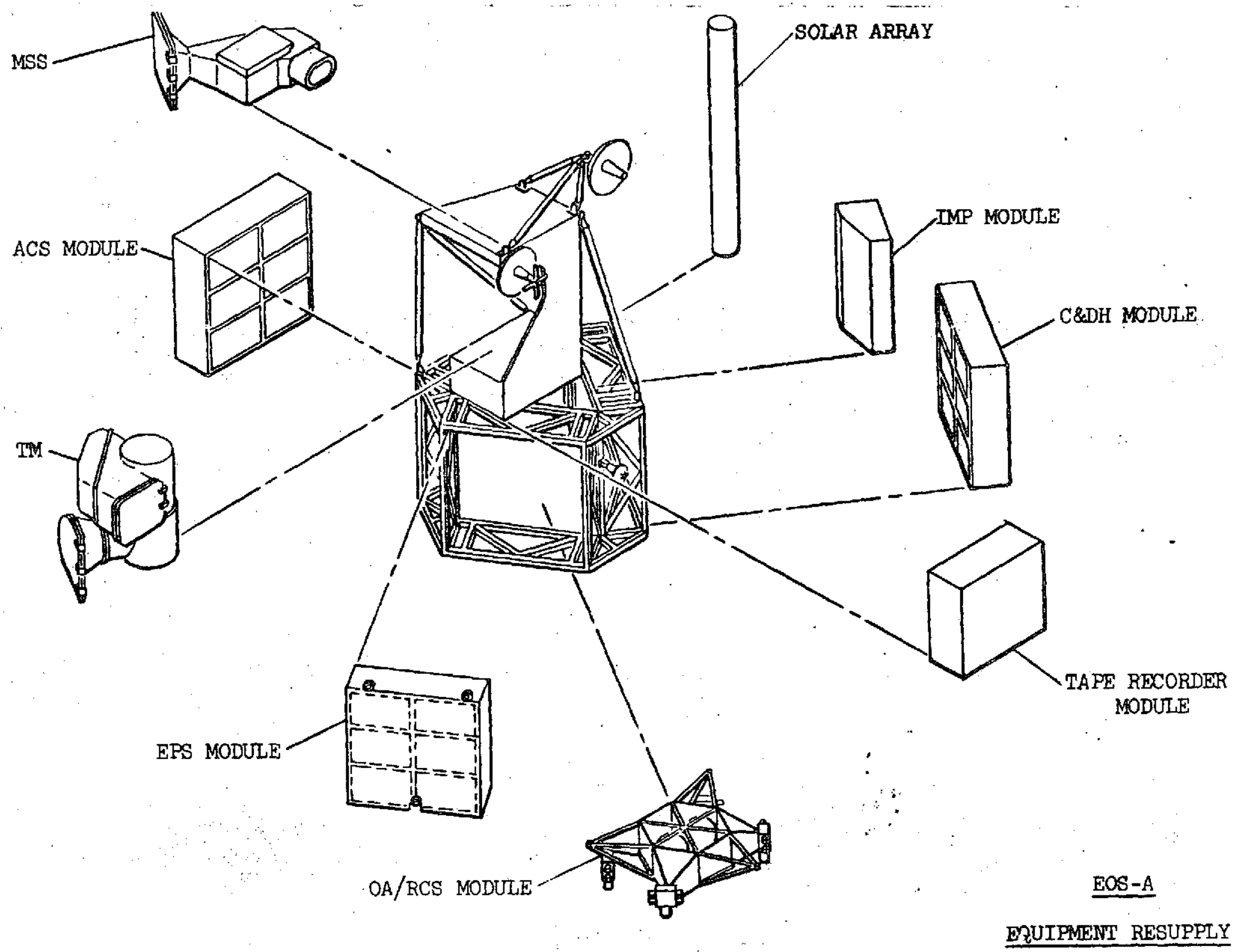


FIGURE 1.1.1-15A

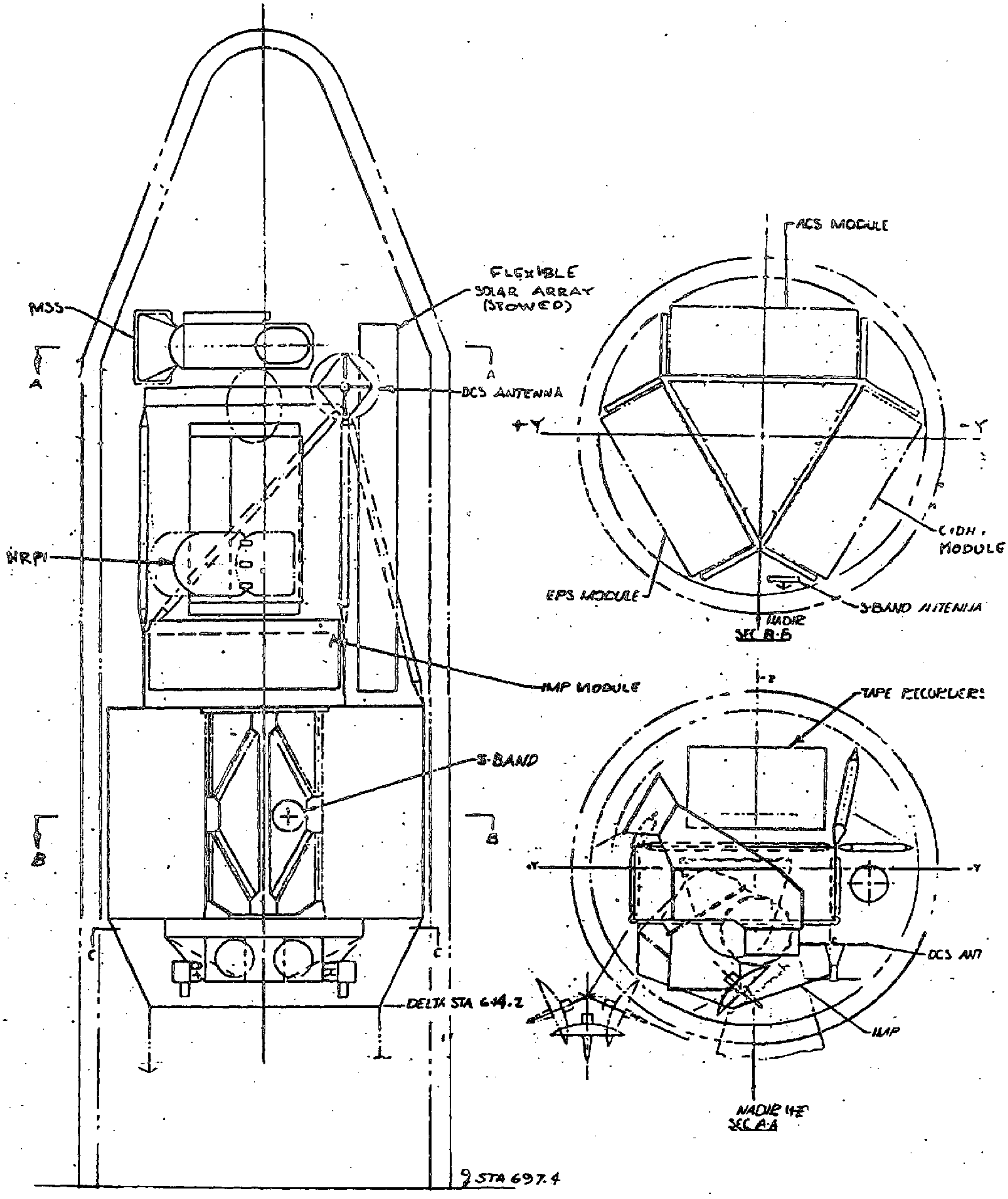


Fig. 1.1.1-16 EOS A' Configuration Delta Launch Vehicle

the spacecraft structure utilizing an additional beam to balance the $\pm X$ loads. The Y, Z loads are balanced by a fitting from array to upper beam.

The tape recorder is located on the -Z side and supported off of the spacecraft structure via a beam or beams to pick up the resupply latches. Removal is in the -Z direction.

The IMP Box is located on the +Z side and is supported similarly to the tape recorder. Removal for resupply is in the +Z direction.

The DCS and X-Band antennas are supported on the upper box beam via appropriate struts.

1.1.1.10.3 EOS B Instrument Section

The Thematic Mapper and the High Resolution Pointable Imager as configured by Hughes can be mounted side-by-side on the triangular spacecraft module support structure as shown in Figure 1.1.1-17. The TM is located on the +Y side to permit a 180° radiator field of view on the shaded side of the spacecraft. The TM scanner is therefore in the forward or +X end with respect to the velocity vector (X axis)avoiding interference with the adjacent HRPI scanner sun shield. The Data Collection System Antenna is located on the Nadir side of the instrument support platform. The steerable X band antenna is deployed from a stowed position below the TM on the Nadir side for maximum earth exposure. The Instrument Mission Peculiar Electronics Module is also mounted on the upper platform central to all instruments and to the Tape Recorder Module which is located above it.

Each of the Instruments, the Mission Peculiar Module and the Tape Recorder Module are removable from the spacecraft using the Grumman Resupply Latching System. The TM and HRPI may be manipulated from the Nadir (+Z) side and the Mission Peculiar Electronics and Tape Recorder Modules from the zenith (-Z) side of the spacecraft.

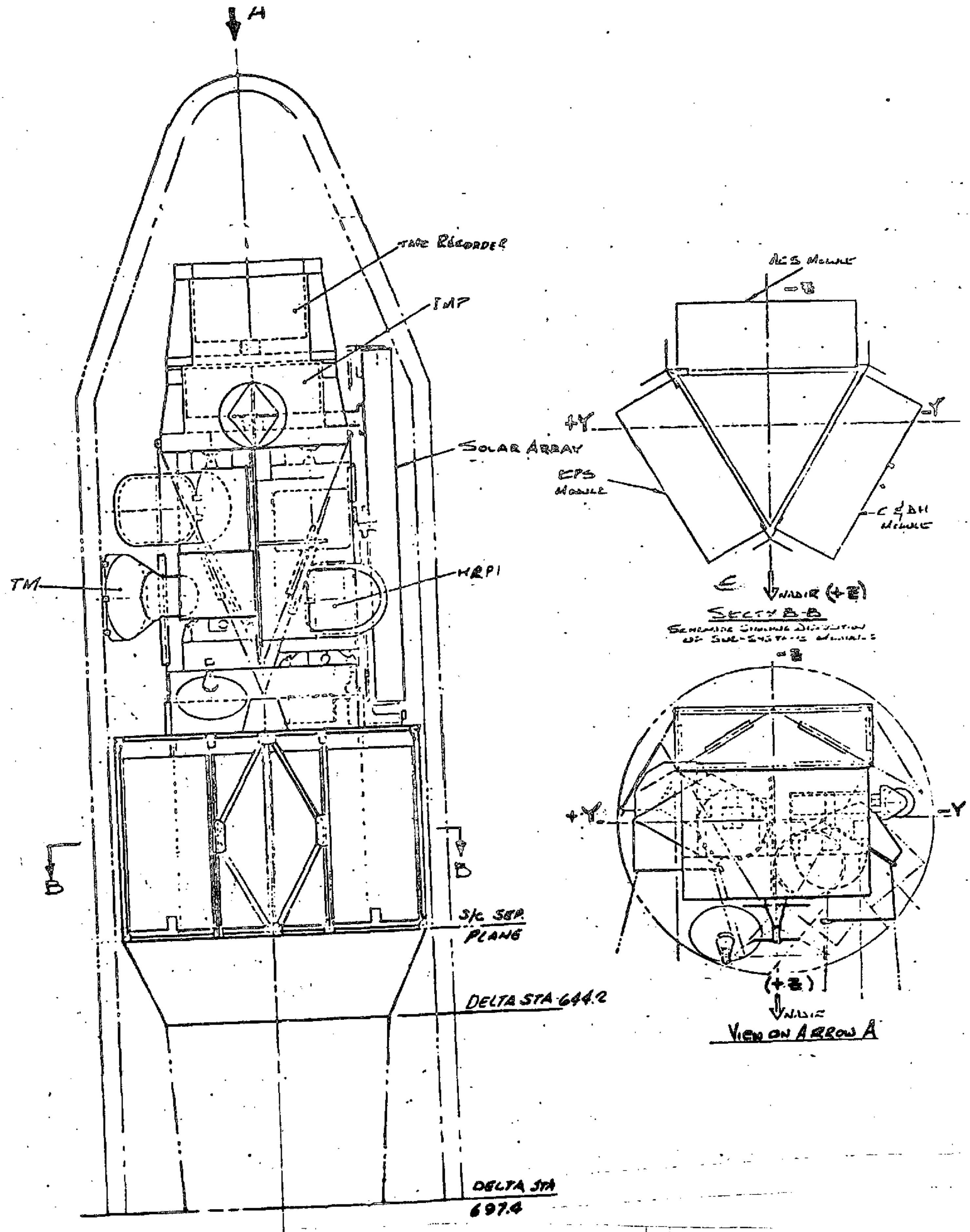


Fig. 1.1.1-17 EOS-B Configuration Delta L/V

The Rollup Solar Array is deployable in the -Y direction, the area of maximum solar energy potential. The actuating mechanism assembly is latch mounted to the forward bulkhead of the spacecraft core structure.
A latching mechanism attached to the instrument platform supports the other end of the stowed array during launch. The Grumman latching system permits replacement of this unit in the -Y direction.

The X-Band Antenna, a 20 inch diameter dish type, is rotatable 62° in any Nadir (+Z) direction and is mounted on the forward face of the lower TM support structure.

The Lower Instrument Support Structure is essentially a 26 inch wide by 51 inch long Beam-Platform, which is attached through the forward bulkhead to the upper caps of the triangular spacecraft structure assembly. The lower latches and track assemblies for the HRPI and TM are attached to the Beam-Platform. The Solar Array Actuating and Latching Mechanisms attached to the interfacing bulkhead are readily accommodated in the hollow interior of the Beam-Platform due to its 18 inch height. Stiffened sheet metal construction with extruded cap members and intercostals are envisioned for this structure.

The upper latches and guides for the major instruments and the array are supported on the underside of a sheet metal Instrument Support Platform 51 inches long, 36 inches wide and 5 inches deep. The X-Band and DCS antenna is attached to a longitudinal beam which is its forward edge. The side face beams and intercostals of the platform provide support for the upper and lower latches of the Mission Peculiar Electronis Module. The former latches are supported on pylon fittings and the latter by the stiffened upper platform face.

Structural continuity between the lower Beam-Platform and the Support Platform is maintained by a central stiffened sheet Beam/Shear Web between the TM and the

HRPI in the X-Z plane. Tripod tubular members support the two rear (-Z) corners of the platform and either connect with interface bulkhead hardpoints or are beamed out to these points. X axis vertical loads are taken out by both the tripods and the vertical beam/shear web. Y and Z axis lateral loads in the platform are reacted in shear by the beam web and by push-pull reactions supplied by the tripods.

The upper latches for the Mission Peculiar Electronic Module are supported on pylon brackets fastened to the top of the support platform and in turn support the Tape Recorder support deck. This $2\frac{1}{2}$ inch deep built-up shelf, 46 inches long and 36 inches wide, mounts the lower Tape Recorder latches and the pylons which mount the upper latches on its stiffened upper surface. Vertical loads are reacted as tension and compression loads in the pylon edge members and lateral loads are sheared out by the pylon bracketry webs to the T/R support deck and instrument support platform.

1.1.1.10.4 EOS C Instrument Section

The instrument section for the EOS-C mission contains the following components as shown in Figure 1.1.1-18. One HRPI, two TM's, one SAR antenna, one tape recorder module, one IMP module, one SAR electronics module, a deployable solar array, a steerable X-Band antenna, a shaped beam X-Band antenna and a DCS antenna.

The structure required to support and house this compliment of components consists of a base support/adapter ring, a rectangular arrangement of beams, 30 inches high, and a truss/beam tower, approximately 17 feet high to support the TM's, SAR, electronics modules, antennas, and solar array.

The base support/adapter ring serves as a base support for the compliment of beams forming the primary support structure. It also serves as the interface/ separation ring when launched in the Titan III B launch vehicle and the interface/

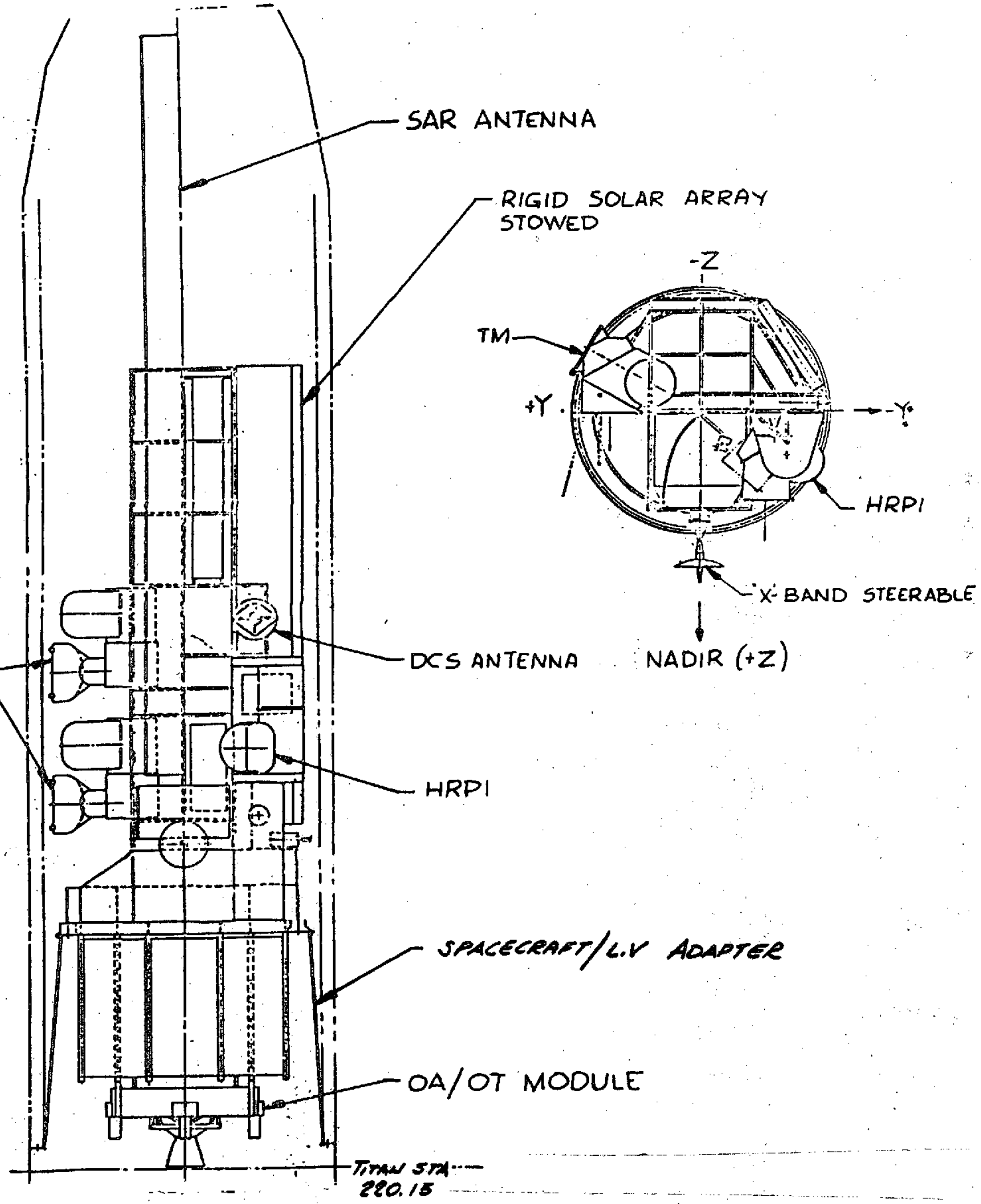


Fig. 1.1.1-18 EOS-C Configuration Titan L/V

support ring when launched in the STS orbiter.

The base support beams consist of three beams parallel to the Z axis, one on the Z axis and two 20 inches either side of the center beam. The ends of the beams terminate at the center of the adapter ring. Joining the ends of the 20 inch beams are two beams running parallel to the Y axis. The center beam terminates on the two transverse end beams. Another transverse beam spans across the adapter ring $6\frac{1}{2}$ inches off the center of the stage on the -Z side providing a center support for the three main beams and the forward (+Z) support for the tower structure. All of these beams are 30 inches high.

Several auxiliary beams extend between the main beams and the base ring to provide support for the subsystem module stage below the instrument stage. Two additional beams extend upward from the main beams to support the HRPI.

The two TM's are mounted one above the other within the tower structure and may be removed laterally in the +Y direction.

The SAR antenna is supported and hinged off the front end of the center tower beam over half the antenna length. The remainder of the antenna is cantilevered from the top of the tower upward.

The tape recorder module is mounted on tracks atop the forward half of the base beam structure. The IMP and SAR electronics packages are mounted within the tower structure.

The solar array is stowed in the -Y, -Z quadrant and consists four 42 x 16 ft panels. They are folded into a 42 inch x 16 ft package and supported off the base beams and the tower structure.

1.1.1.10.5 <u>Delta Configuration - 2 MSS Instrument Section</u>

The configuration shown in Figure 1.1.1-19 mounts two MSS instruments, a Tape Recorder module and an IMP with attendant antennae, on top of the conven-

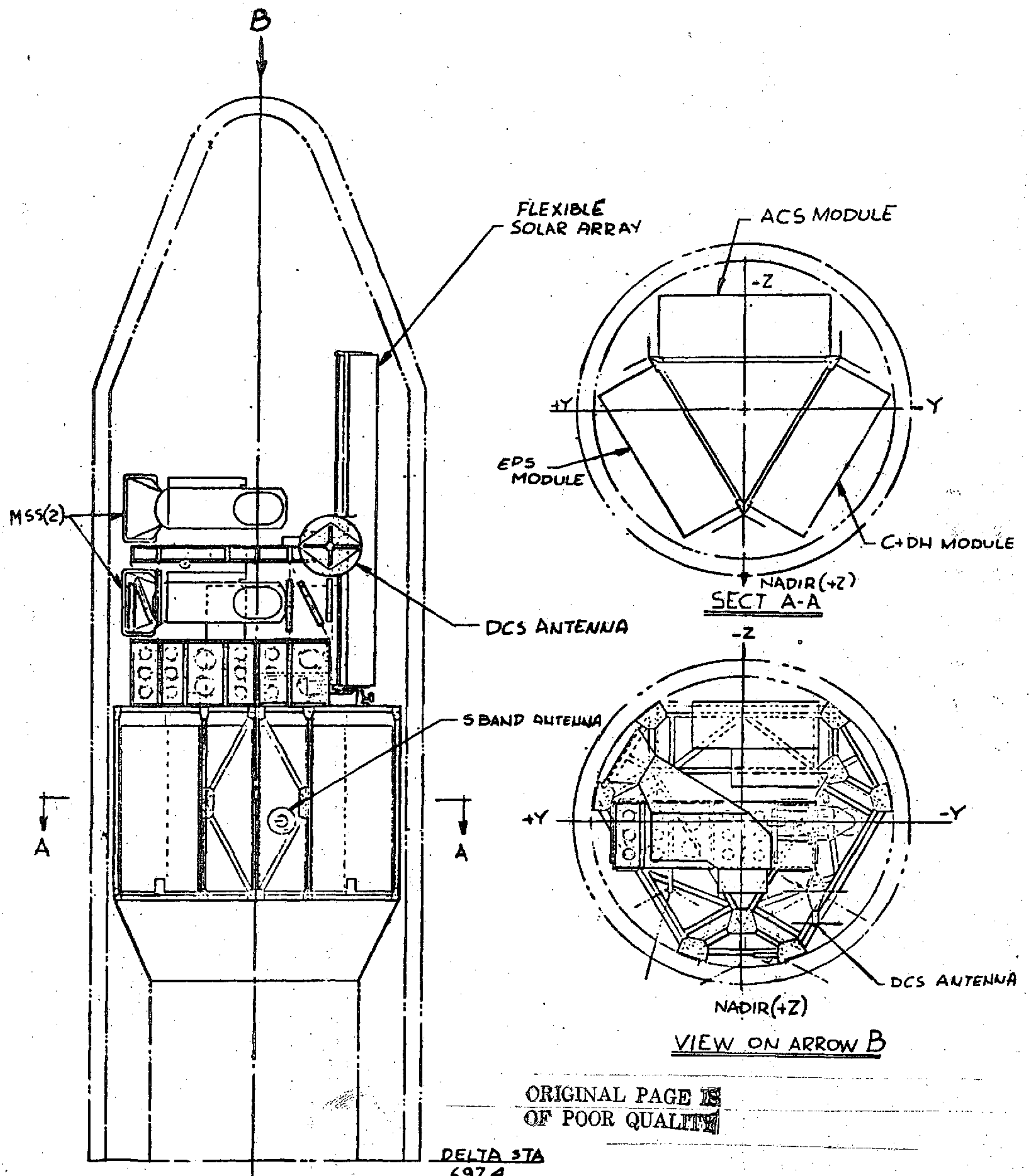


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Fig. 1.1.1-19 EOS Configured for ERTS Mission (2 MSS) Delta L/V tional subsystems modules arrangement. A 20 inch deep box beam of aluminum alloy shear webs and stringers is mounted to the upper bulkhead of the spacecraft core structure at four points where it intersects the three vertical webs of that structure. To this box beam is mounted one of the MSS instruments together with the Tape Recorder Module and the IMP. It should be noted that to remove this MSS instrument, it is necessary first to remove the tape recorders and IMP.

The second MSS instrument is mounted on a deck which is supported by struts off the box beam. This MSS is positioned vertically above the lower MSS but can be removed without first removing another component.

Both MSS are attached by three latches to their supporting structure.

Antennae positioned above the spacecraft core structure are mounted off the box beam or the upper deck.

Fields of view requirements for the instruments and antennae are fully satisfied in this arrangement.

This configuration utilizes the flexible solar array which is described in Paragraph 1.1.1.7.3. Mountings for this array are provided at the lower edge of the box beam and the upper deck. The box beam incorporates a tunnel to accept part of the solar array mechanism.

1.1.2 STRUCTURAL AND DYNAMIC ANALYSIS

1.1.2.1 Introduction

This section presents the results of the structural studies performed to define the member sizes which meet the vehicle design requirements. The most significant drivers in sizing the structural members are the stiffness required to meet the launch vehicle design frequencies in both the lateral and longitudinal directions. The selected configurations, both baseline and preferred, for the Delta and Titan Launch vehicles were evaluated for stiffness requirements. The structural idealization used to estimate the stiffness of each structural arrangement was based on an evaluation of primary load paths, effectivity of structural members and estimated sizes for the preferred configurations. It should be noted that these analytical studies are preliminary in scope and further analyses are required to obtain more definitive results. The preferred configurations structural members were sized, as noted above, for stiffness; estimates were made of member sizes for given candidate materials and the fundamental frequencies evaluated. Where the structure did not meet the frequency requirements, the members were resized. Conservative mass distributions were used in all cases. It will be noted throughout this analysis that the weights used are higher than those finalized in the weights section. Since these design weights were selected early in the study, it was decided to use higher values to avoid significant changes in the results. The steady state load cases were used to size members where the design loads were critical, although, the basic structure is stiffness critical. The study included an evaluation of the following structural materials: Aluminum alloys, Titanium alloys, Beryllium, Beryllium/aluminum alloy and composites.

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1.1.2.2 SUMMARY

The results of these preliminary structural studies indicate the following:

- o Delta Preferred - the requirements for strength, stiffness and weight allocation are satisfied for:
  - Spacecraft designed in aluminum alloy
  - Instrument payload structures using composite tubular trusses or support beam of aluminum. The matrix of all potential instrument combinations should be examined further.
  - Equipment module using three point support system, aluminum honeycomb bulkhead and aluminum side trusses.
  - Shuttle installation system using six hard points to mate in cradle in the Orbiter payload bay.
  - Elimination of transition ring.
- o Titan Configurations - studies show that the baseline and preferred meet design requirements.
- o Composite materials and beryllium/aluminum alloy can be effectively used in specific applications to reduce weight.
- o A summary of the preliminary estimated frequencies are given in Table 1.1.2.2-1.

TABLE #.1.2.2-1

SUMMARY OF CALCULATED FREQUENCIES

CONFIGURATION	Weight Lbs		Frequency - Hz	
	Payload	Spacecraft	Long.	Lat.
Titan Baseline				
First estimate of longitudinal stiffness	2120	5610	20	-
Second estimate of longitudinal stiffness	2120	5610	33	-
Transition Ring/Adapter restrained	2120	5610	-	14
Titan Preferred				
Case I Stiffness	1500	4000	-	17
Case I Stiffness	3700	4000	-	12
Case II Stiffness	1500	4000	-	22
Case II Stiffness	3700	4000	-	15
Delta Preferred				
Case I TM, MSS, DCS & Solar Array	817	2124	-	14
Case I TM, MSS, DCS & Solar Array	817	2124	34	
Case II 2 MSS & Solar Array	690	2220	33	-
Equipment Module				
Longitudinal			70	-
Lateral - 1" honeycomb .020" facesheets			-	79

1.1.2.3 Titan Baseline Configuration:

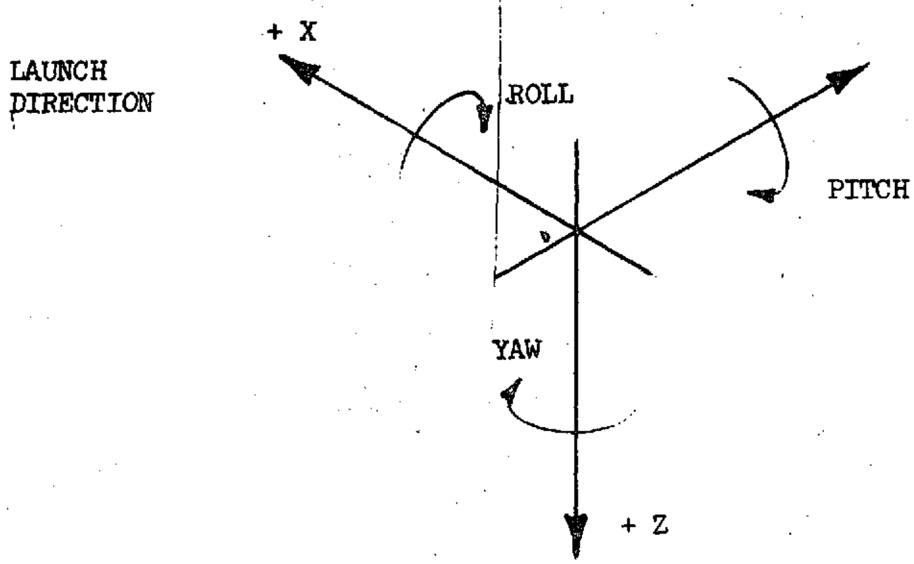
The basic structural arrangement is taken from the GSFC and Mega reports. The design gross weight used for this analysis is as follows:

	Wt (LBS)
Structure	2134
Subsystem Modules	729
Orbit Transfer Syst.	1817
Thermal Control	150
Solar Array	240
Wiring	190
Steerable Antenna	40
Contingency	1060
	6060
Instrument Payload	1277
	7337
Adapter	390
Launch Weight	7727

TITAN/SHUTTLE BASELINE DESIGN ULTIMATE LOADS

TABLE 1.1.2.3-1

LAUNCH	Fx	Fy	Fz
o Titan			
- Lift-off	-25312	±22000	or ±22000
- Main engine cutoff	-118860	±16500	or ±16500
o Shuttle			
- Liftoff	-25312	±3300	-8800
- Orbiter end burn	-36318	±2200	-5500
- Entry	±2750	±5500	33000
- Landing	±16500	±16500	27500
- Crash (1)	69700	0	0
(2)	0	0	33000



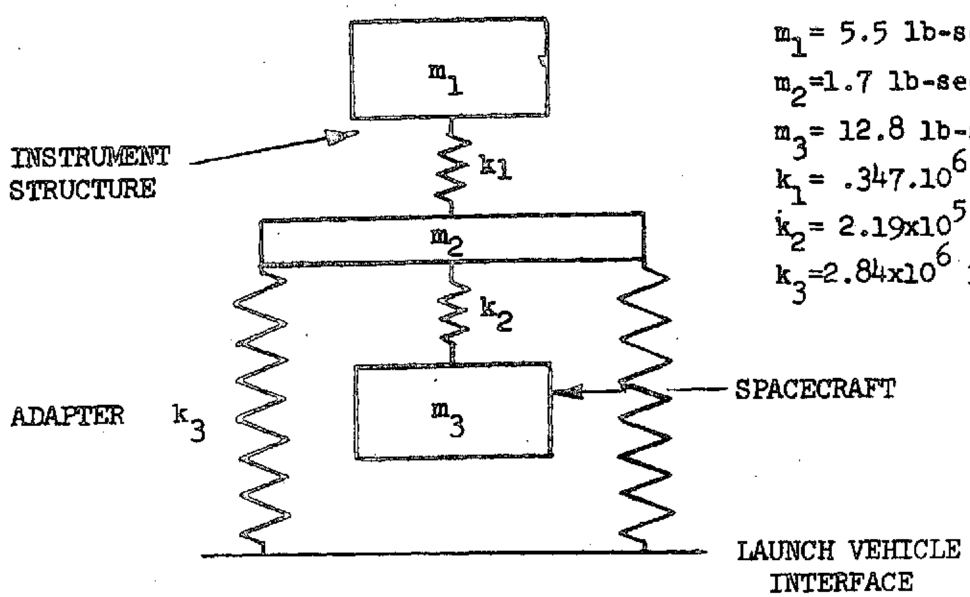
COORDINATE SYSTEM

Stiffness Evaluation

The preliminary evaluation of the longitudinal stiffness and frequency of the Titan baseline structure was based on two different assumptions:

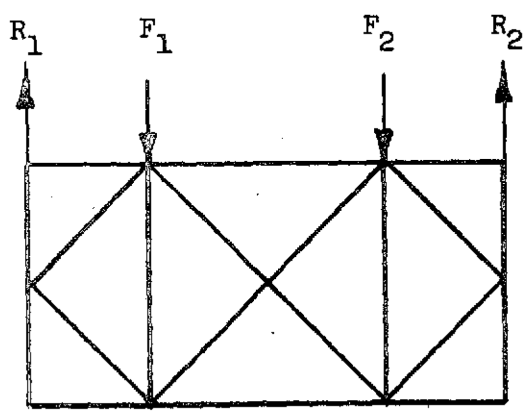
1. The spacecraft I sections which support the module tracks act as cantilever beams with zero slope at the centerpost cross bracing; the module load applied nine inches from the adapter support reaction. This assumption gave a longitudinal stiffness of $k = 2.19x10^5$ lbs/in. The instrument structure was sized for a total approximate weight of 2000 lbs with a center of gravity of 68 inches above the transition ring. The longitudinal stiffness is $3.47x10^5$ lbs/in. The adapter was sized for launch loads using a one inch thick honeycomb fabricated of aluminum alloy with .020 inch face sheets and a core density of 2.8 pcf. The member was analyzed as a cylinder 102 inches long and average diameter of 115 inches. This design resulted in a $k=1.47x10^6$ lbs/in. However, in the frequency analysis it was decided to use the value given in Mega report for $k=2.84x10^6$ lbs/in.

The three mass-spring dynamic system idealization, shown below, was analyzed resulting in a fundamental longitudinal frequency of 20 Hz.



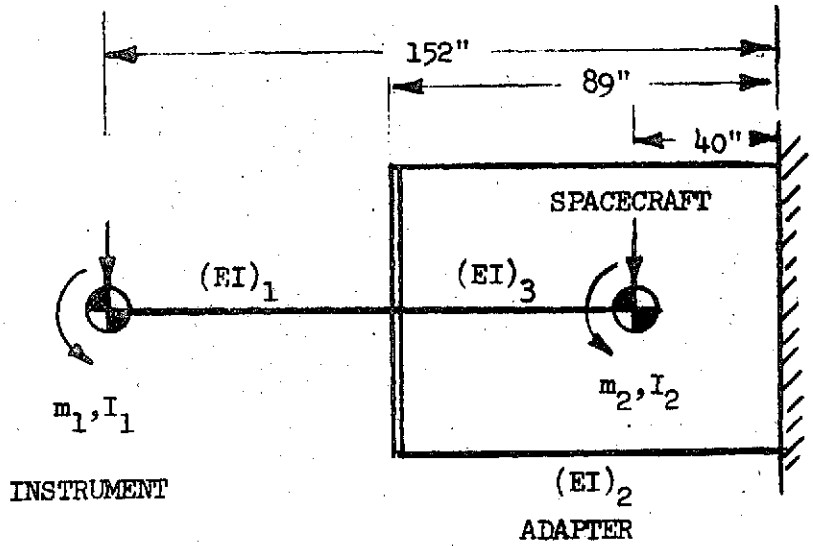
$m_1 = 5.5$ lb-sec^2/in
$m_2 = 1.7$ lb-sec^2/in
$m_3 = 12.8$ lb-sec^2/in
$k_1 = .347 \cdot 10^6$ lb/in
$k_2 = 2.19 \times 10^5$ lb/in
$k_3 = 2.84 \times 10^6$ lb/in

2. In the second structural simulation the spacecraft was idealized as a truss as shown below:



R_1 and R_2 are reactions at the adapter, F_1 and F_2 are the loads applied by the equipment modules. It is assumed that the equipment modules act as truss structures in the four bays around the center structure. Each truss forms the criciform structure which carries the spacecraft equipment modules and in turn is supported at the adapter thru the transition ring. It is conservatively assumed that the module inertia forces are applied at the vertical post members as shown by the forces F_1 and F_2. Calculation of the truss stiffness shows that $k= 7.23 \times 10^5$ lbs/in. This value was used with the stiffness of the adapter and instrument structures given above together with the masses to establish the longitudinal dynamic system resulting in a fundamental longitudinal frequency of 33 hz.

The two mass-spring dynamic system idealization, shown below, was analyzed resulting in a fundamental lateral frequency of 14 Hz.



$m_1 = 5.9$ lb-sec ^{2}/in

$m_2 = 14.1$ lb-sec ^{2}/in

$I_1 = 13.75 \times 10^3$ lb-in-sec^2

$I_2 = 19.3 \times 10^3$ lb-in-sec^2

$(EI)_1 = 1.4 \times 10^{10}$ lb-in^2

$(EI)_2 = 22 \times 10^{10}$ lb-in^2

$(EI)_3 = 4.3 \times 10^{10}$ lb-in^2

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The results of the longitudinal frequency evaluations for assumptions (1) and (2) above show that the required stiffness exists in the designed structure based on the assumed simple models. A more detailed finite element model which incorporates realistic structural member idealization is required to give a better definition of the stiffness and frequency.

1.1.2.4 Titan Preferred Configuration

The Titan Preferred configuration is based on the GAC design with the following gross weight:

Instrument Payload & Struct	1500	lbs
Spacecraft	3850	lbs
Adapter	150	lbs
	5500	lbs

DESIGN ULTIMATE LOADS

TABLE 1.1.2.4-1

VEHICLE	X	Y	Z
Titan	-89100	±12375 or	±12375
Shuttle			
o End Orbiter Burn	-27275	± 1650	- 4125
o Entry	± 2062	± 4125	+24750
o Landing	±12375	±12375	20650
o Crash	49500	0	0
	0	± 8250	0
	0	0	24750
	- 8250	0	0
	0	0	11000

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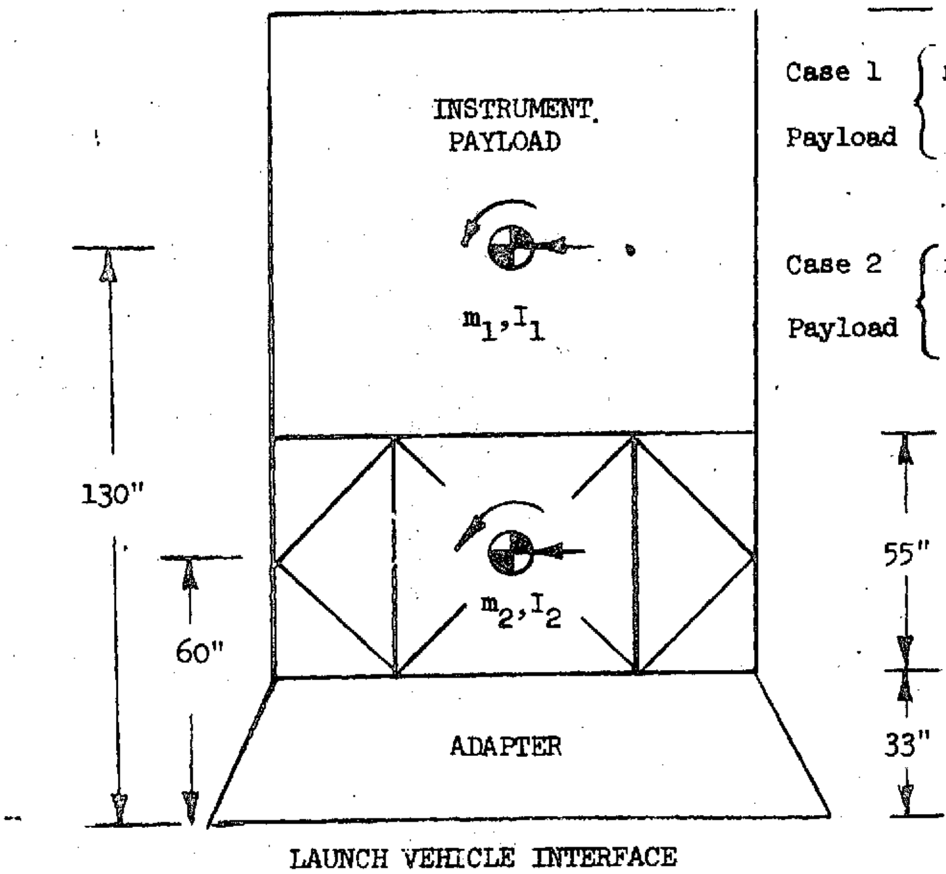

Case 1 Payload $\begin{cases} m_1 = 3.89 \text{ lb-sec}^2/\text{in} \\ I_1 = 9.48 \times 10^3 \text{ lb-in-sec}^2 \end{cases}$

Case 2 Payload $\begin{cases} m_1 = 9.59 \text{ lb-sec}^2/\text{in} \\ I_1 = 23.38 \times 10^3 \text{ lb-in-sec}^2 \end{cases}$

SPACECRAFT

$m_2 = 10.36 \text{ lb-sec}^2/\text{in}$

$I_2 = 14.95 \times 10^3 \text{ lb-in-sec}^2$

The structure was sized for each case and the moments of inertia are plotted on the attached curves given in Figure 1.1.2.4-1

The stiffness and mass data were used in a two mass-spring dynamic system idealization, shown above, resulting in the fundamental lateral frequencies given in Table 1.1.2.4-2.

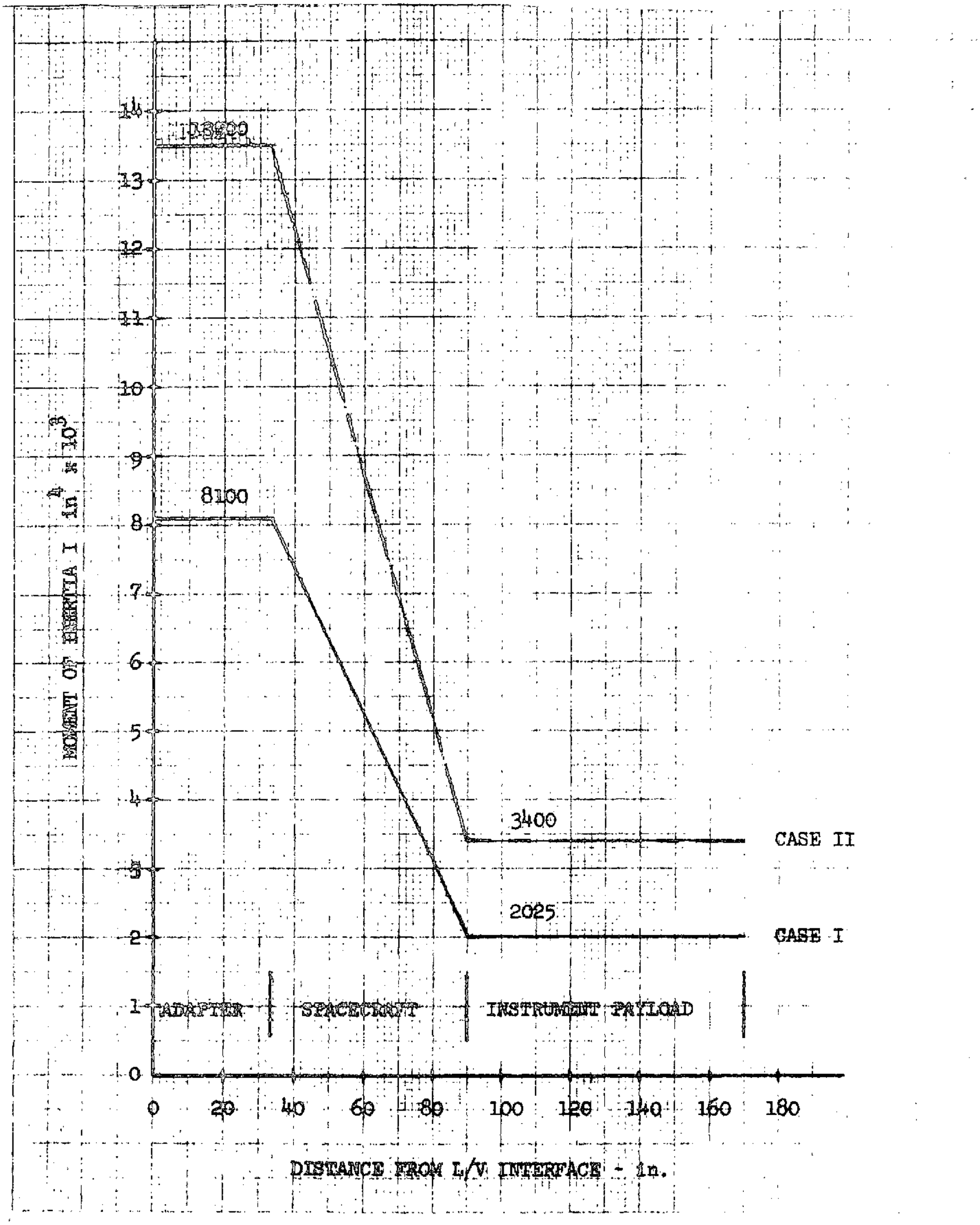

Fig. 1.1.2.4-1 TITAN III Preferred GAC Configuration Estimated Moment of Inertia

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TRADE STUDY REPORT NO.

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Table 1.1.2.4-2 TITAN III Preferred Lateral Frequency

Stiffness		Payload Weight		Total Weight	Frequency Hz
I	II	1500 lbs	3700 lbs		
X		X		5500	17
X			X	7700	12
	X	X		5500	22
	X		X	7700	15

1.1.2.5 Delta Preferred Configuration

The Delta Preferred Configuration for the basic spacecraft is defined in the GAC drawings shown in paragraph 1.1.1.4. The primary structure consists of three vertical shear webs forming a triangular cross section core vehicle; extending from the webs are six vertical trusses which form the support for the three equipment modules. The equipment modules are supported at three points as shown on the drawings in paragraph 1.1.1.7. In this arrangement, primary structural loads are not induced in the equipment modules. The design goal is to configure the structural arrangements of all possible Instrument Payload Structures such that they are attached at the hard points of the basic core structure of the spacecraft, which in turn attaches at six hard points to the adapter stiffeners.

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DELTA/SHUTTLE PREFERRED CONFIGURATION

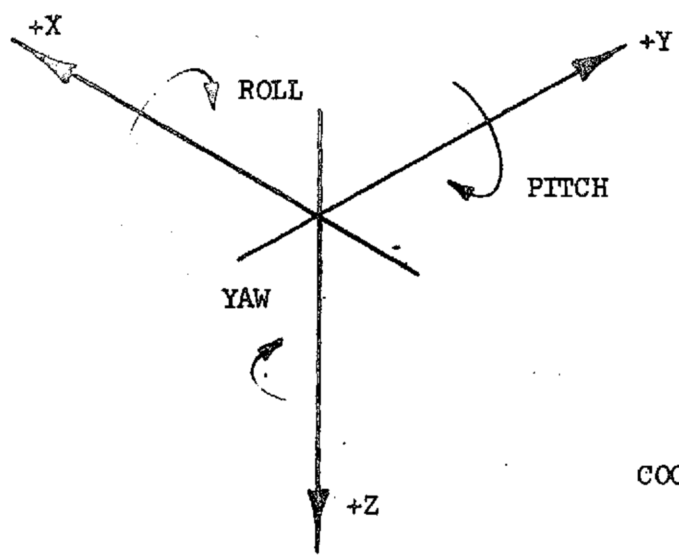
Design Weight 2941 lbs (Note: this weight is conservative but the results of the analyses are not significantly effected).

Delta/Shuttle Design Ultimate Loads

TABLE 1.1.2.5-1

Launch Vehicle	Fx	Fy	Fz
Delta Engine Cutoff	-53700	±8730 or	± 8730
Shuttle			
- Lift-off	-10150	±1320	-3530
- Orbiter End Burn	-14600	± 880	-2200
- Entry	±1100	±2200	13230
-Landing	± 6620	±6620	11000
- Crash	26500	0	0
	0	0	13230

LAUNCH DIRECTION



COORDINATE SYSTEM

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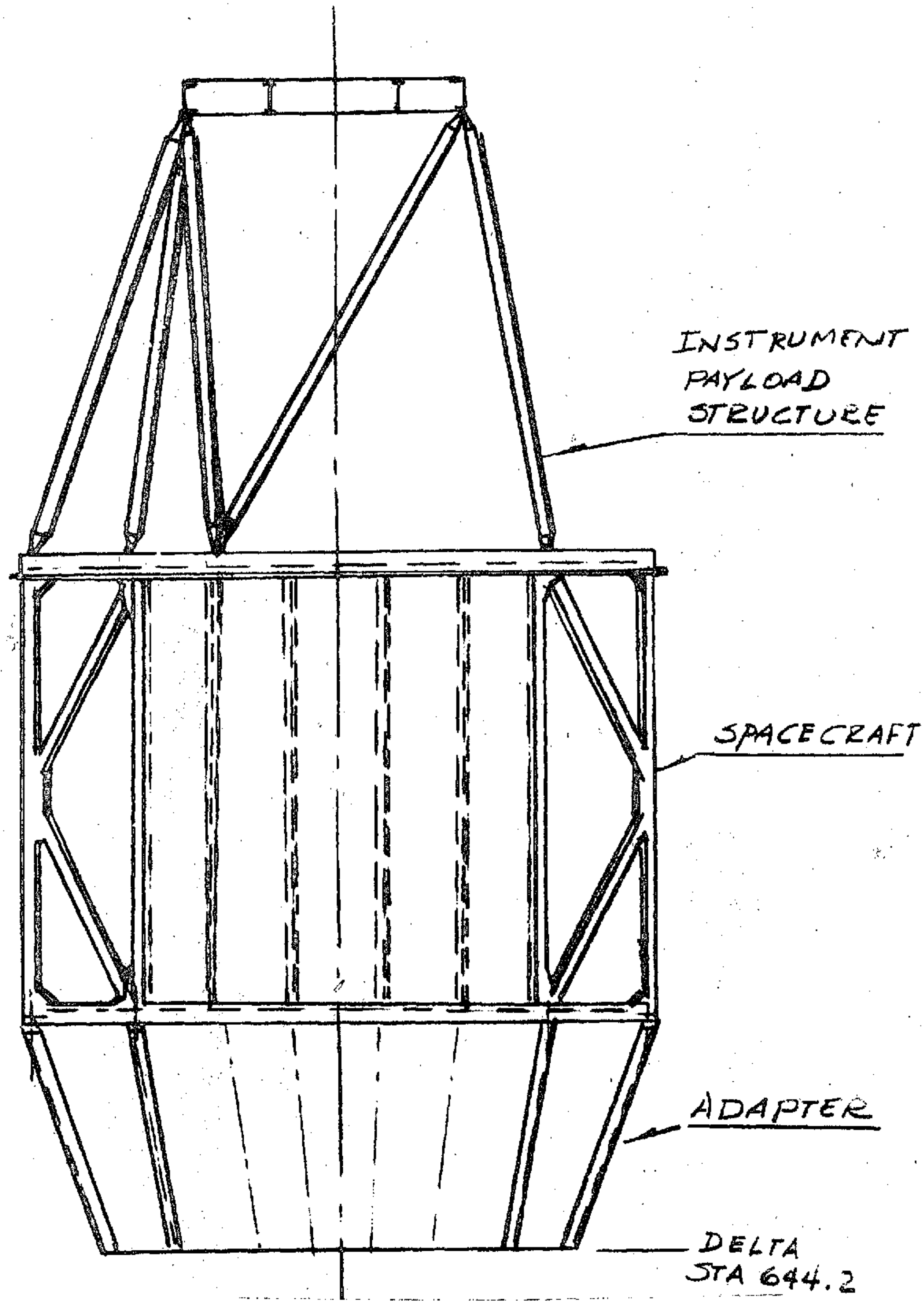


Fig. 1.1.2.5-1 Delta Preferred Configuration with Typical Instrument Payload Structure

Typical Arrangement

The combined Instrument payload structure, spacecraft and adapter structures were sized for the design requirements for two separate payload components and support structures. The first Instrument Payload/Structure included the following: TM, MSS, DCS and Solar Array; the combined weight equal to 817 lbs including structure. The support structure was a Boron/Epoxy tubular truss which attached to hard points on the core vehicle. The second instrument structure analyzed included 2 MSS instruments and the solar array all supported on a beam structure attached to the core structure.

Analysis of Longitudinal and Lateral Stiffness

o Spacecraft/Adapter Stiffness

Spacecraft and adapter wgt 2124 lbs- Instrument payload and structure 817 lbs.

Design load in spacecraft tubes at the six hard points for the Delta launch condition is - 14100 lbs ultimate.

Tube size 2 inches by 083 - Aluminum Alloy

$$\text{Area} = .4999 \text{ in}^2$$
$$\sigma_c = -28200 \text{ psi}$$
$$F_c = -30380 \text{ psi}$$
$$M.S = +0.08$$

All the tube members on the spacecraft structure are sized to the equivalent rectantular section with an area of .4999 in^2. These values are used to calculate the longitudinal and vertical stiffnesses. The longeron area of the adapter is 0.60 in^2. The calculated stiffness for spacecraft and adapter structure for a single mass system with center of gravity at the center of the spacecraft is:

$$k = 3.54\text{x}10^5 \text{ lbs/inch}$$

o Instrument Payload Structure Stiffness/Frequency

- Truss structure case I, longitudinal

The truss structure analyzed for strength and stiffness is shown on the following figures. The platform is supported at the four corners MDLH by tubes MA, MB, MC; DD', DE, DF; LK, HK, HJ. The tube loads were estimated for the launch condition; however, the loads were too low to give sizes which would meet the stiffness requirement. The structural sizes were recalculated several times to satisfy requirements. The tubes sizes were all 2 inches x.065 inches except HJ and LK which were estimated to be 3 inches x.095. The material used was Baron/Eppxy with the following ply orientation: 25% $\pm 45^{\circ}$, 60°, 0°, 15% 90°. This lay up gives a modulus of elasticity of 20×10^6 psi. To obtain the sizes for tubes in other materials, the areas are ratioed by the inverse ratios of Youngs Modulus. The structural arrangement discussed above has a calculated longitudinal stiffness:

$$k = 9.65 \times 10^5 \text{ lbs/in}$$

o Truss Structure Case I, Lateral

- Assumptions:

1. All lateral stiffness in $\pm$Z direction provided by Truss JH, HF, FD (neglect induced torque which is resisted by trusses in $\pm$ Y direction).
2. MSS wgt 220# applied at upper platform.
3. TM wgt 350# equally divided between upper platform and upper deck on S/C structure
4. 175# Solar array on upper platform
5. Assume I of S/C struct 50% effective lateral

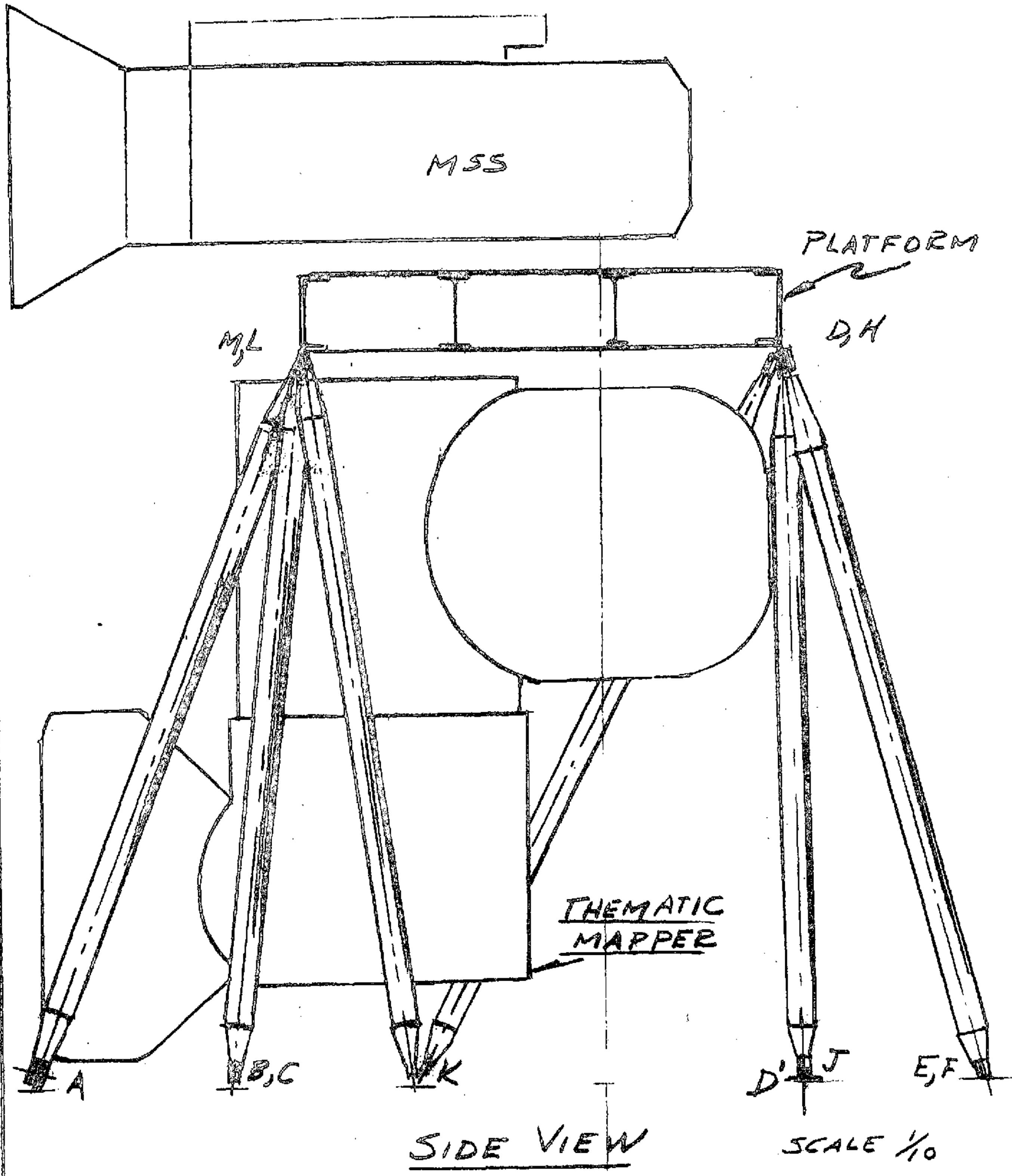
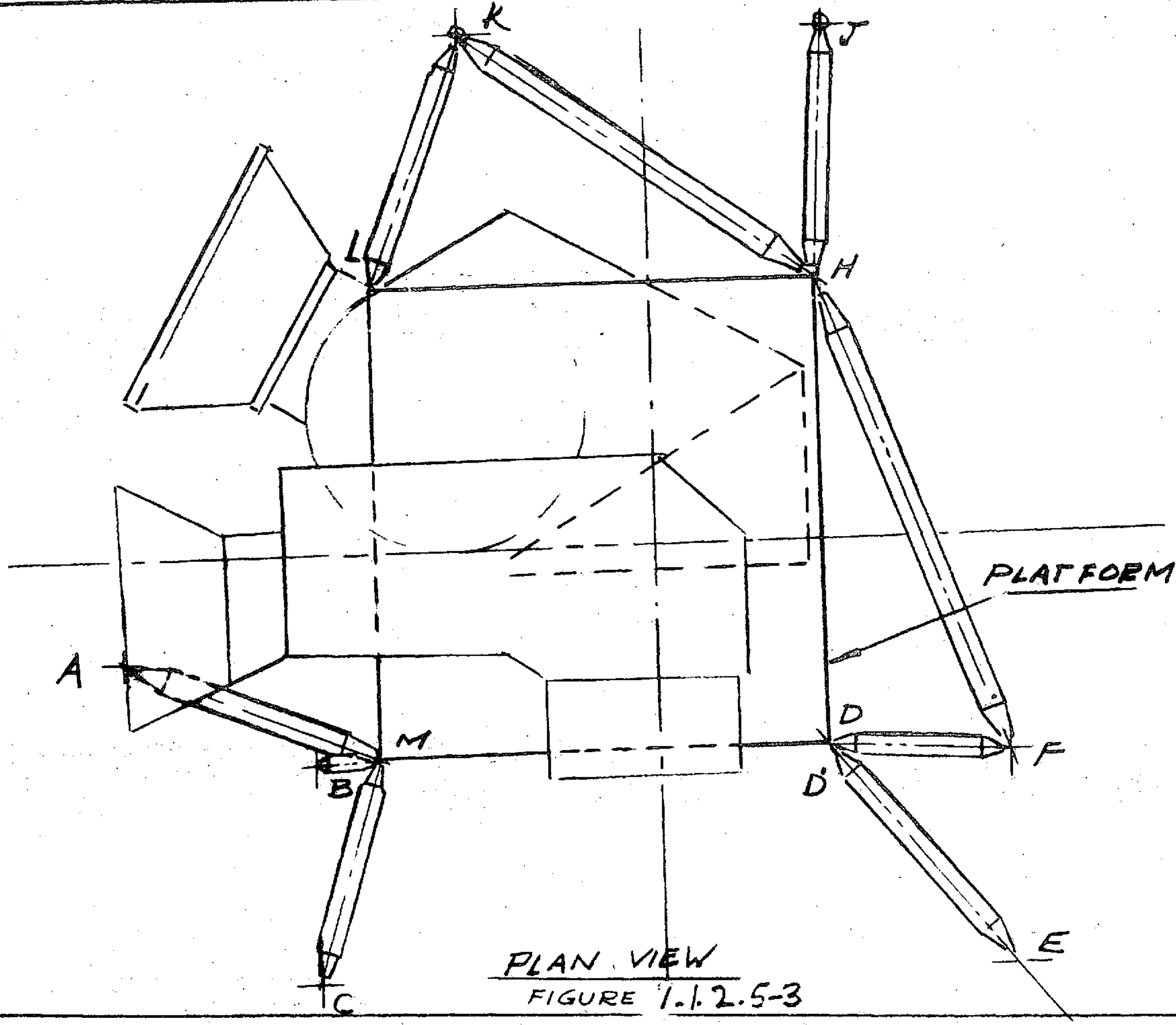


Fig. 1.1.2.5-2 Support Structure- Instrument Payload Delta L/V



PLAN VIEW
FIGURE 1.1.2.5-3

TITLE	TRADE STUDY REPORT NO.
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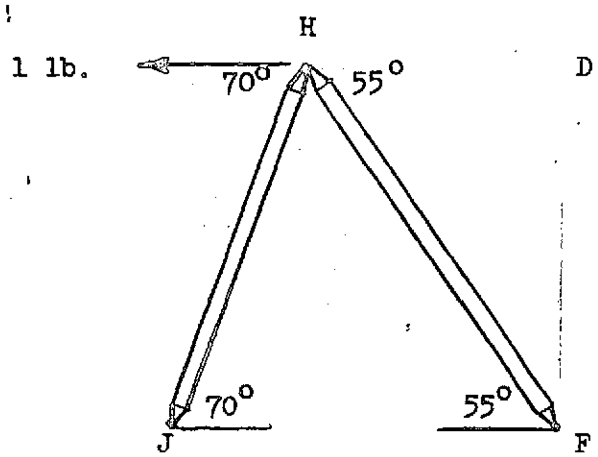
LATERAL FLEXIBILITY OF JH, HF (approximate)



L_{HF} = 63 in

L_{HJ} = 55 in

Deflection in lateral direction due to one lb load

Tube 2" OD x 065 $B/_{E_p}$

$A = .395 \quad E = 20 \times 10^6$

MEMBER	L	AE	$\frac{L}{AE}$	U	U^2	$\frac{U^2 L}{AE}$
HF	63	7.9×10^6	7.97×10^{-6}	1.15	1.3225	9.1655×10^{-6}
HJ	55	7.9×10^6	6.96×10^{-6}	-1	1.00	6.96×10^{-6}

$\Sigma\ 1.613 \times 10^{-5}$

$\delta = 1.613 \times 10^{-5}$ in/lb

$k = 6.20 \times 10^4$ lb/in

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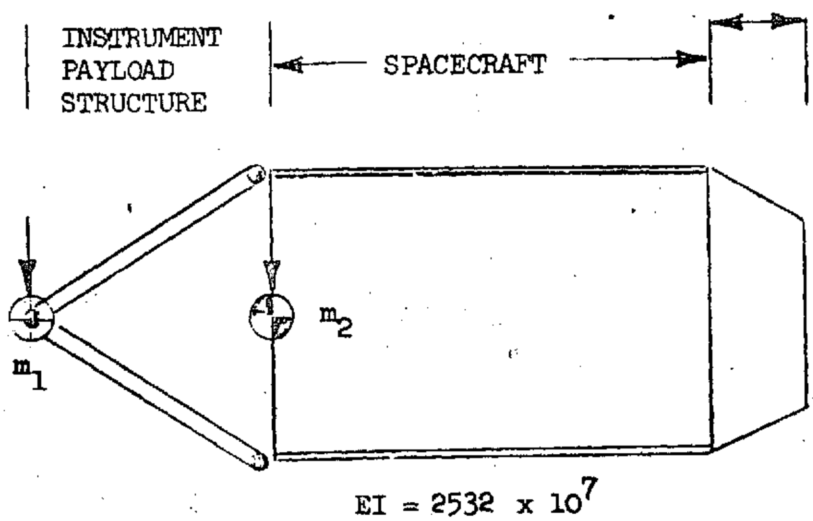


m_1 = 1.61 lbs-sec $^2/$in

m_2 = 5.94 lbs-sec^2/in

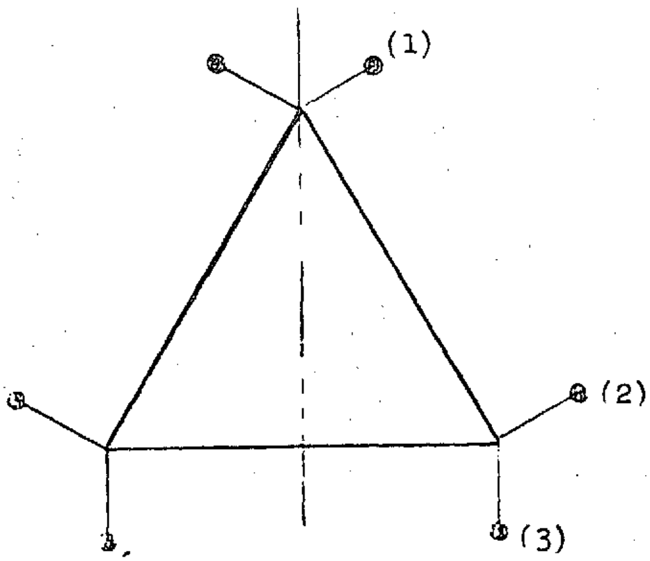
W = 2910#

L_1 = 59 in

L_2 = 90 in

Total m = 7.54

Since truss HJ, HF, DF, DE is on one side, use only I of areas (1), (2), (3)



AREA OF MEMBERS = .5 in^2
(1), (2), (3)

ELEMENT	A	y	Ay	$\bar{y}$	$A\bar{y}$	Ay^2
(1)	.5	38	19			722
(2)	.5	- 8	- 4			32
(3)	.5	-32	-16			512
	1.5		- 1			Σ = 1266

$I = 2532 \text{ in}^4$

$$\alpha_{11} = \delta + \frac{L_2^3}{3EI} + \frac{L_1 L_2^2}{EI} + \frac{L_1^2 L_2}{EI} = 5.336 \times 10^{-5}$$

$$\alpha_{21} = \alpha_{12} = \frac{L_2^3}{3EI} + \frac{L_1 L_2^2}{2EI} = 1.824 \times 10^{-5}$$

$$\alpha_{22} = \frac{L_2^3}{3EI} = .960 \times 10^{-5}$$

$$\left[K\right] = \begin{bmatrix} 5.336 & 1.824 \\ 1.824 & .960 \end{bmatrix}^{-1} \times 10^{-5} = \begin{bmatrix} .5345 & -1.0156 \\ -1.0156 & 2.9710 \end{bmatrix} \times 10^{5} \text{ lb/in}$$

$$\omega^4 - \left(\frac{K_{11}}{m_1} + \frac{K_{22}}{m_2} \right) \omega^2 + \frac{K_{11} K_{22} - K_{21}^2}{m_1 \; m_2} = 0$$

ω = 87.8 , 274.8 rad/sec

f = 14 , 43.7 Hz

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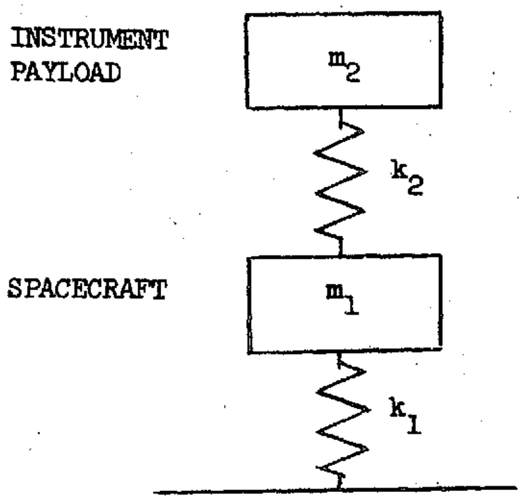
o <u>Truss Structure Case I</u> Longitudinal



$m_2 = 2.1$ lbs-sec^2/in

$k_2 = 9.65 \times 10^5$ lbs/ in

$m_1 = 5.44$ lb-sec^2/in

$k_1 = 3.54 \times 10^5$ lbs/in

$$\omega^2 = \frac{k_1 + k_2}{2m_1} + \frac{k_2}{2m_2} \pm \sqrt{\frac{1}{4}\left(\frac{k_1 + k_2}{m_1} + \frac{k_2}{m_2}\right)^2 - \frac{k_1 k_2}{m_1 m_2}}$$

$f_1 = 34$ Hz fundamental longitudinal frequency

$f_2 = 129$ Hz

o **Beam Structure Case I Longitudinal**

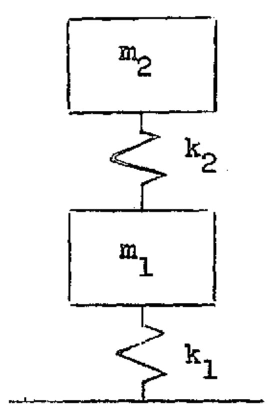
The instrument support structure consists of a box beam as shown on the attached sketch 20 inches deep and supported at an average 30 inch span by the core vehicle vertical beam webs. The following are the components and weights:

2 MSS Instruments	440 lbs
Solar Array	175 lbs
Structure	75 lbs
	690 lbs

The beam component sizes include 0.30 in^2 cap areas, 0.040 webs and .032 cover skin. The longitudinal stiffness is:

$k = 1.92 \times 10^6$ lb/in $\quad m = 1.80$

SPRING-MASS SYSTEM



$m_2 = 1.8$ lbs-sec^2/in

$k_2 = 1.92 \times 10^6$ lb/in

$m_1 = 5.7$ lbs-sec^2/in

$k_1 = 3.54 \times 10^5$ lb/in spacecraft

$$\omega^2 = \frac{k_1 + K_2}{2m_1} + \frac{k_2}{2m_2} \pm \sqrt{\frac{1}{4}\left(\frac{k_1 + k_2}{m_1} + \frac{k_2}{m_2}\right)^2 - \frac{k_1 k_2}{m_1 m_2}}$$

$= 212, 1192$ rad/sec

$f = 33$ Hz 180 Hz

1.1.2.6 EQUIPMENT MODULE STRUCTURE

The equipment module structure as shown in the drawings of paragraph 1.1.1.6. A major part of the equipments are mounted on the honeycomb face; the side and horizontal trusses around the sides of the module provide stiffening and support attachement loads.

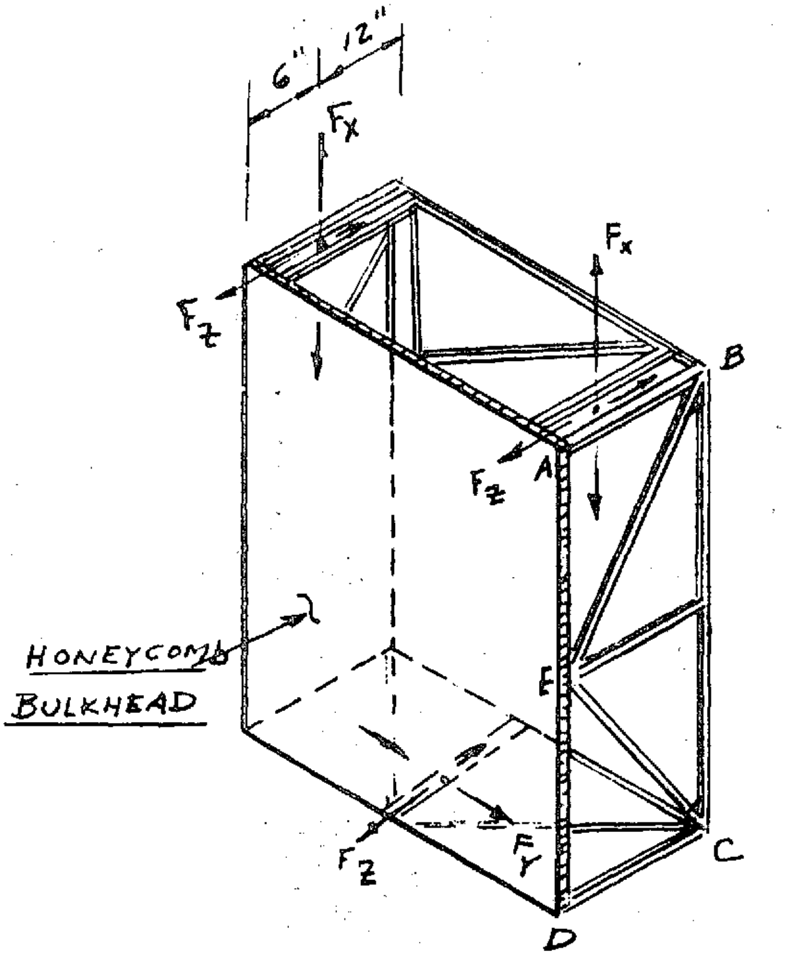
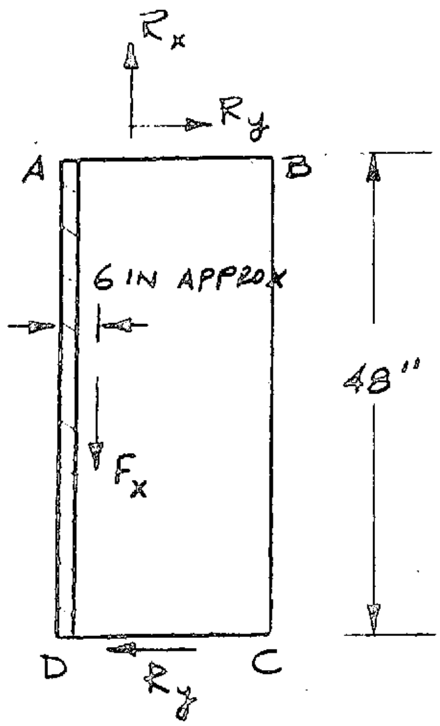


FIGURE 1.1.2.6-1

EQUIPMENT MODULE

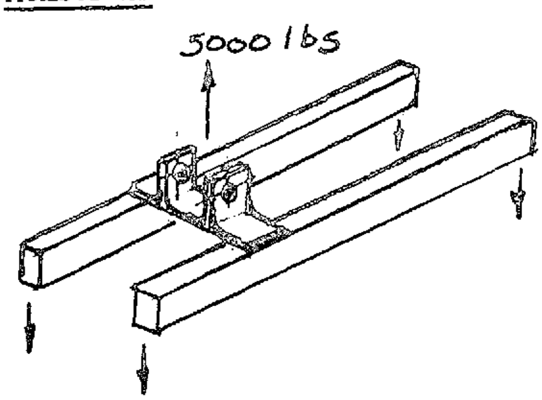
Total weight 400 lbs attached to the honeycomb bulkhead. Longitudinal load factor 25 ultimate. Lateral load factor 15 ultimate.



For longitudinal frequency the primary flexibility is caused by bending in members AB and axial deformation in AD.

Fx = 400 x 25 = 10000 lbs ultimate

<u>Member AB</u>

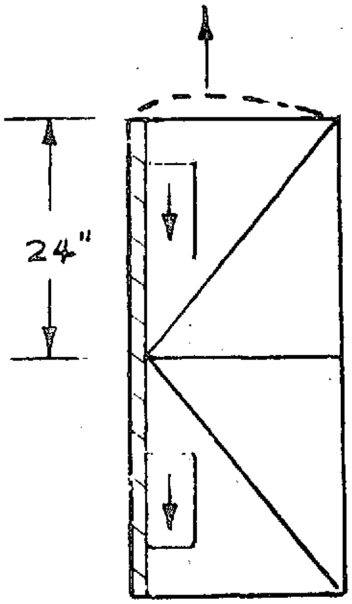


size of square tubes 2 inches x .10 inch aluminum alloy

stiffness assuming load applied at fitting and reacted at ends of beams

$$k = 1.04 \times 10^5 \text{ lb/in}$$

Vertical Truss Member



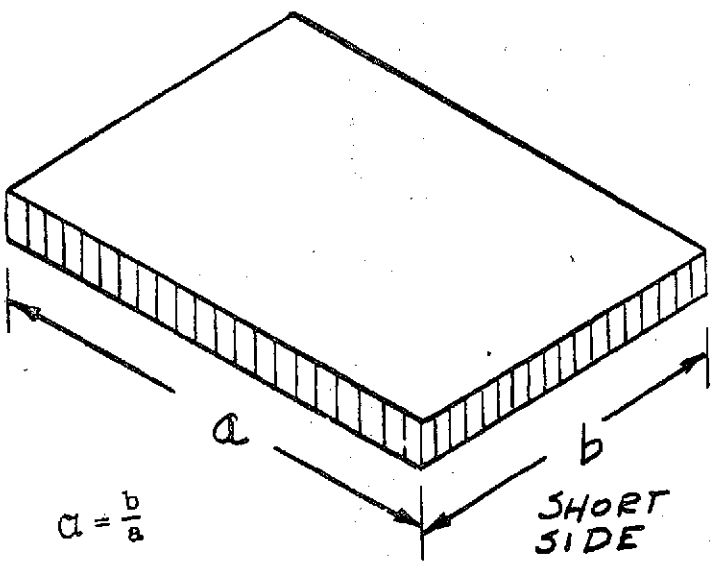
Assume total mass supported at end of 24" bar on forward bulkhead with area .35 in^2.

$$k = 2.9 \times 10^5 \text{ lb/in}$$

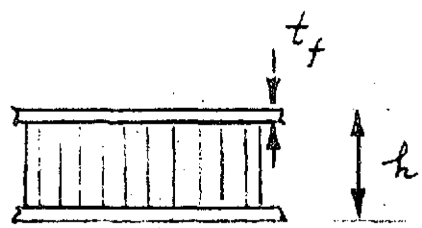
$$\frac{1}{k} = \frac{1}{k_1} + \frac{1}{k_2}$$

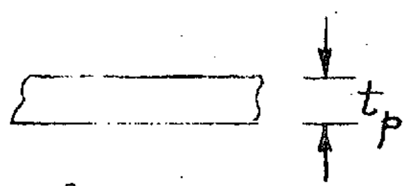
$$f = \frac{1}{2\pi}\sqrt{\frac{K}{m}} \quad = \quad 70 \text{ hz longitudinal}$$

Frequency Evaluation of Simply Supported Honeycomb Bulkhead



$$\alpha = \frac{b}{a}$$





$$t_p^{\,3} = 6\ t_f\ h^2$$

$$t_p = (6\ t_f\ h^2)^{1/3}$$

$$\Delta = \frac{0.1422\, W_p\, b^4}{E\, t_p^{\,3}\,(1 + 2.21\,\alpha^3)}$$

W_p = Wgt intensity (of equivalent flat plate)

$$W_p = \frac{abt_p\rho}{ab} = t_p\rho \ \text{lb/in}^2$$

$$\Delta = \frac{0.1422\rho b^4}{E\, t_p^{\,2}\,(1 + 2.21\alpha^3)}$$

for $E = 10^7$ lb/in^2

$\rho = 0.1$ lb/in^3

$h = 1$ in.

$t_p = 1.82\ t_f^{\,1/3}$

$t_p^2 = 3.31\ t_f^{\,2/3}$

$$\Delta = \frac{1.422\, b^4 \times 10^{-9}}{t_p^{\,2}\,(1 + 2.21\,\alpha^3)}$$

$$\Delta = \frac{1.422\, b^4 \times 10^{-9}}{3.31\, t_f^{\,2/3}\,(1 + 2.21\alpha^3)}$$

$$f = \frac{3.834}{\sqrt{\Delta}}$$

reference Machine Design Sept. 1971

$$f_1 = \frac{3.839}{\sqrt{\Delta}}\sqrt{\frac{W_p}{W_e}}$$

$W_e = 386/48^2 = .168$ lb/in^2

Frequencies are calculated and given in Table 1.1.2.6-1 for three face sheet thickness.

TABLE 1.1.2.6-1

HONEYCOMB CONFIGURATION	FACESHEETS IN.	FREQUENCY f	f_1
48x48 Honeycomb devided into two	.032	110	63
rectangular 24x48 panels	.025	102	56
a = 48, b = 24	.020	94	50
48x48 Honeycomb devided into four			
square 24x24 panels	.032	175	100
a = b = 24	.025	161	89
	.020	149	79

Note: the frequencies are plotted in the Figure 1.1.2.6-2 which follows
f is frequency of unloaded panel
f_1 is frequency of loaded panel

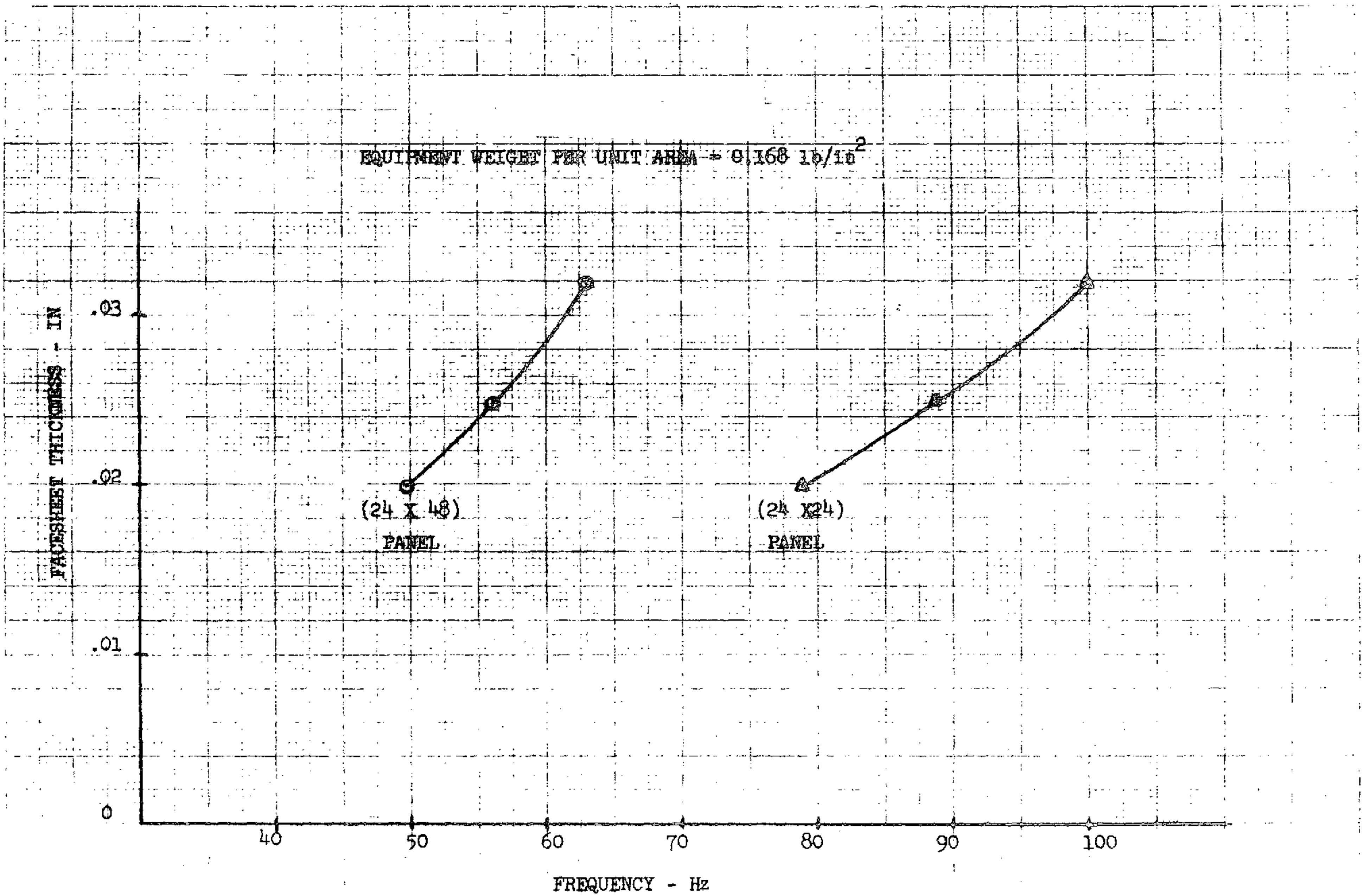


Fig. 1.1.2.6-2 Honeycomb Panel

1.1.2.7 Candidate Materials

The Table 1.1.2.7-1 summarizes the significant properties of the candidate materials which have application to the spacecraft structure. Most of the structural sizing was based on aluminum alloy properties either 6061 or 2219 except the tubular structure used to support the Instrument payload. In these applications the higher modulus made the Baron/Epoxy more competetive weight wise. Since stiffness is a major driver in structural sizing, estimating the effects of selecting other materials than those calculated in sizing can be done by ratioing the areas by the ratio of Young's Modulus.

1.1.2.8 Analysis of Delta 2910 Fairing as a Load Carrying Member

Design Loads:

Total weight 2941 lbs

Limit Load Factors

$$\left.\begin{array}{l} \eta_x = 12.3 \\ \eta_{lat} = .65 \end{array}\right\} \text{engine cutoff}$$

$$\left.\begin{array}{l} \eta_x = 2.9 \\ \eta_{lat} = 2.0 \end{array}\right\} \text{liftoff}$$

NOTE: The Fairing should also be checked for the aerodynamic and inertial forces at max $q\alpha$. These data are not available.

Ultimate Design Loads

Engine Cut-Off	F_x = 55260 lbs
	F_{lat} = 2900 lbs
Lift-Off	F_x = 12800 lbs
	F_{lat} = 8823 lbs

$$N_x = \frac{M}{\pi r^2} + \frac{F}{2\pi r}$$ maximum longitudinal axial load per incy in cylinder

N_x = 231 lb/in Engine Cut-off

N_x = 192 lb/in Lift-off

TABLE 1.1.2.7-1

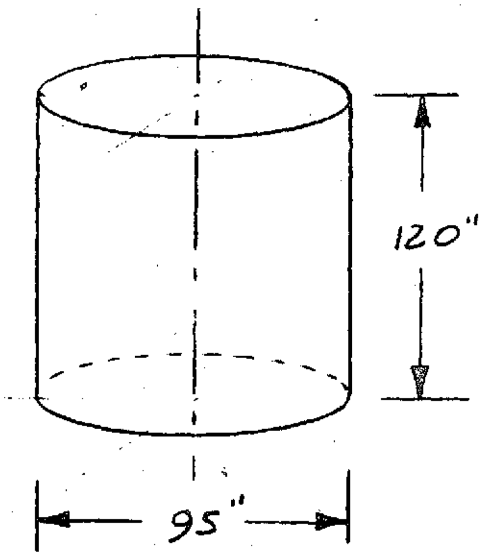
MATERIALS PROPERTIES - CANDIDATE MATERIALS

MATERIAL	ρ lbs/in^3	Ftu Ksi	Ftu/ρ X10^{-6}	Fty Ksi	E 10^3 ksi
o Aluminum Alloy					
2219-T81	.102	62	.61	46	10.5
2219-T87	.102	63	.62	51	10.5
6061-T6	.098	42	.43	35	9.9
o Titanium					
6 AL-4 V ANN	.16	134	.84	126	16.
6 AL-4 V STA	.160	160	1.0	145	16
o Beryllium-Aluminum					
Be - 38% AL	.076	54.3	.71	41	31
o Berryllium Sheets	.067	65	.97	42	42.5
Shapes	.067	40	.60	27	42.5
o Composite					
Boron/Epoxy	.073	115	1.58	-	20
Graphite/Epoxy	.056	105	1.88	-	12
Hybrid	.075	100	1.33	-	20

NOTES: (1) All properties are at RT

(2) Composite properties based on crossply lay up of 60% @ 0°, 25% @ ±45°, 15% @ 90° any other layups will give other properties.

Fairing Dimensions:



Candidate Materials - Properties

	Boron/Epoxy	Graphite/Epoxy (UHM)	Beryllium-Aluminum Alloy
ρ lbs/in^3	.073	.056	.076
E (0°) psi	30×10^6	25×10^6	--
E* psi	12×10^6	10×10^6	31×10^6
E*/ρ	1.64×10^8	1.7×10^8	4.08×10^8

* Note: These properties are for the following lay up of plies:

35% 0°, 30% $\pm 45^\circ$, 35% 90°

Failure mode of cylinder is compressive instability.

Critical buckling stress is given by:

$$\sigma_{crit} = 0.6 \ \gamma E \ {}^{t}/_{r} \qquad \text{where } \gamma = 1 - 0.901 \ (1 - e^{-\phi})$$

$$\phi = \frac{1}{16}\sqrt{\frac{r}{t}} \quad \text{for } \frac{r}{t} < 1500$$

Reference NASA SP 8007

1. σ_{crit} for Graphite/Epoxy using t = 0.072

$$\sigma_{crit} = 3273 \text{ psi}$$

$$\sigma_x = \frac{231}{.072} = 3208 \text{ psi}$$

2. σ_{crit} for Be-Al (Lockalloy) using 0.050

$$\sigma_{crit} = 5874 \text{ psi}$$

$$\sigma_x = \frac{231}{.050} = 4620 \text{ psi}$$

3. Weight for Shell skin only

$$W_{GR/EP} = 157 \text{ lbs}$$

$$W_{Be\text{-}Al} = 136 \text{ lbs}$$

1.1.2.9 Trade Study - COMPARISON OF RELATIVE SHELL STRUCTURAL WEIGHTS FOR VARIOUS MATERIALS. DELTA FAIRING

$$\sigma_{allowable} = 0.6\,\gamma E \frac{t}{r}$$

$$\gamma = 1 - 0.901\,(1 - e^{-\phi})$$

$$\phi = \frac{1}{16}\sqrt{\frac{r}{t}}\,,\ \text{for}\ \frac{r}{t} < 1500$$

$$\sigma_{applied} = \frac{Nx}{t}$$

$$\sigma_{allowable} = \sigma_{applied}$$

$$t = \left(\frac{N\,x\,r}{0.6\,\gamma E}\right)^{1/2}$$

For Design condition: $r = 47.5$ in, $N_x = -231$ lb/in

Does not include max $q\alpha$ condition

Material	E	ρ
Aluminum	10.5	.1
Titanium	16	.16
Be - Al	31	.076
B/Ep	12	.073
Gr/Ep	10	.056

The function γ, the knock down factor, versus r/t is taken from figure 2 of SP 8007 and superimposed on the curves calculating t from $t = \left(\frac{N_x r}{0.6\gamma E}\right)$ with variable γ which are function of r/t for each material.

Figure 1.1.2.9-1 shows the relative weight function $t\rho$, shell thickness required times density of material, for various candidate materials. The Beryllium-Aluminum alloy is the lightest with the composites also showing a competetive weight potential.

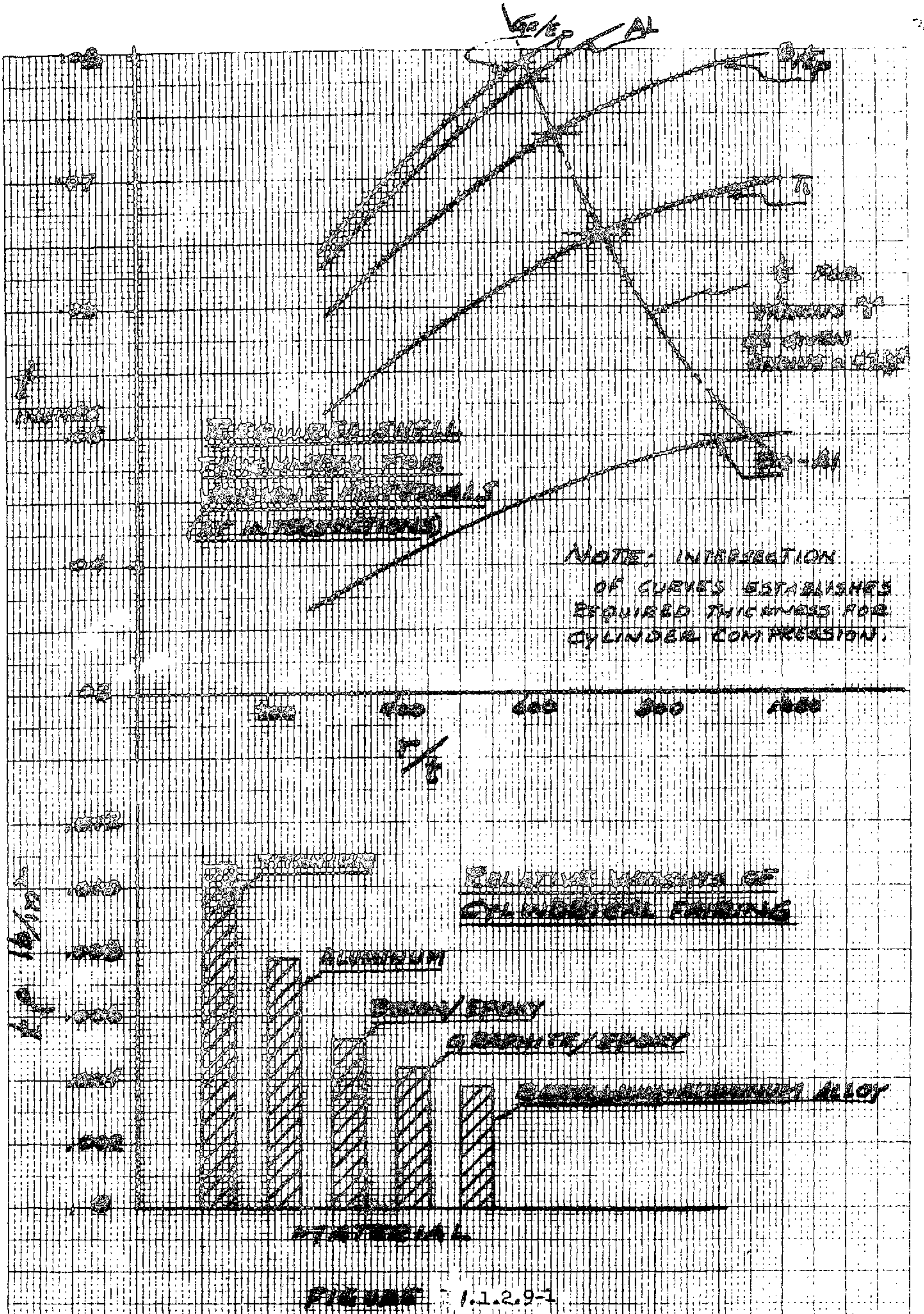


FIGURE 1.1.2.9-1

1.2 Spacecraft Thermal Control

1.2.1 Summary

Several structural concepts have been considered for both the Delta and Titan approaches. The compliment of instruments being evaluated adds further mission peculiar complexity. As a result, the buildup and use of a detailed comprehensive thermal model, at this point in time, was impractical. Therfore, each section of the structure for an available Delta configuration was evaluated separately (i.e., instrument structure, transition ring, module structure, orbit adjust stage). The intent of this analysis was to evaluate thermal control requirements and determine minimum cost-weight approaches.

The baseline design was assumed to be a 70°F structure temperature with an insulation effective emittance equal to .05. Heater power as a function of structure temperature and insulation effective emittance was evaluated.

In support of this study a detailed orbital heat flux study was conducted. Transient and orbital average heat fluxes were generated using a thirty (30) surface model. The range of altitude and DNTD considered for the EOS Land Resources Missions were evaluated to obtain minimum and maximum heat fluxes. The impact of the solar array emission to spacecraft surfaces was included.

The details of the structure thermal analysis are discussed in section 1.2.4 and are summarized as follows:

- Reductions in structure heater power from pre-study estimates have been achieved by structure/thermal design approaches that minimize external surface area and maximize the use of multilayer thermal insulation. Deletion of thermal skins in the instrument areas and substitution of insulated trusses and decks result in significant reduction in weight, heater power and cost.

- For a baseline requirement 70°F structure and insulation effectiveness of .05 the total structure heater power is 66 watts. Using an insulation effectiveness of .02, which should be readily achievable, reduces the heater power to 28 watts. Reducing the structure temperature to 40°F decreases the heater power requirements to the range of 15-38 watts (range of insulation effectiveness). Although 100 watts of structure heater power was assumed for preliminary solar array sizing, it is apparent that the total structure heater power penalty will be less than 40 watts.
- Preliminary feedback from the instrument contractors indicate concurrence with a thermally decoupled design interface and therefore acceptance of lower structure temperatures. Maintaining the transition ring at 70°F should be only a transient condition, during contact periods. A module support structure of 40°F is consistent with the minimum anticipated equipment operating temperatures. A 40°F OAS structure is consistent with minimum propellant temperature requirements.

1.2.2 Basic Cost Data

The costs of thermal control hardware were compiled from vendor quotes and in-house manufacturing estimates. In addition other significant thermal cost items such as solar array impact and test time were evaluated. The unit costs for thermal control hardware are given in Table 1.2-1 .

Table 1.2-1
Thermal Control Hardware Costs

Item	Non-Recurring Cost	Unit Cost	Source
Louver	$8,100.	$11,700	Fairchild Hiller
Var. Cond. Heat Pipe	$15,000 - 20,000	$5,000-6,000	GAC
Isothermalizer Heat Pipe	$10,000-15,000	$4,000-5,000	GAC
Solid State Thermostat	$46,300	$417	Cox & Co.
Bi-Metallic Thermostat	$4,600	$90	Sunstrand
Heaters	———	$10	Minco
Skins (OAO Type)	———	$500/FT^2	GAC
Insulation	———	$80 - 180/FT^2	GAC

For known approaches or qualified hardware, a single unit cost has been given. For items with potential development or different approaches, a range of unit costs have been given. Review of the unit cost data in Table 1.2-1 and consideration of the quantity of hardware required for a program with several spacecraft indicates the following:

- o The use of variable conductance heat pipes as opposed to louvers is a more cost effective approach for reducing heater power. In addition, better temperature control (reduced range) and higher heat rejection capability is achieved with heat pipes.

- o The use of a solid state thermostat is comparable in cost to a quad - redundundant bi-metallic thermostat approach.

- The use of skins is an expensive approach and should be minimized. In addition significant weight is saved (approximately .26 $^{lb}/ft^2$).

- The cost estimate for insulation is based on the simplified OAO approach (i.e. no mock-up or built-in structural requirements). This approach will readily yield an effective emittance in the range of .02 to .03.

The impact for heater power in terms of solar array costs are as follows:

honeycomb array	750/watt
roll-up array	750-1750/watt

The reduction in module acceptance testing costs for a $70^{\circ}F \pm 10^{\circ}F$ design versus a $70^{\circ}F \pm 50^{\circ}F$ is 16K per module.

1.2.3 Orbital Heat Flux

Transient and orbital average heat fluxes were generated for the range of sun-synchronous orbit parameters covering the EOS LRM mission. External heat fluxes consisting of direct solar, earth albedo and earth IR, were determined using the orbital heat flux program. Secondary effects such as blockage of albedo and earth IR radiation and reflection of direct solar, albedo and earth IR were not considered. IR emission from the solar array to the spacecraft was not considered in determining the particular orbit at which the worst case flux occurs; however, solar array fluxes to the appropriate spacecraft modules were evaluated at the worst case orbit condition and added to the external fluxes to establish thermal extremes for the design of each subsystem module. A total of 14 computer runs were made for both earth-oriented spacecraft surfaces and sun oriented solar array surfaces for the following thermal environment constants and combinations of orbit parameters:

Thermal Environment Constants

Solar Constant: Vernal Equinox - 430 BTU/HR FT^2
Winter Solstice - 444 BTU/HR FT^2
Summer Solstice - 415 BTU/HR FT^2

Albedo Constant: .30

Earth Emission: 75 BTU/HR FT^2

Orbit Parameters

Orbit Altitudes: 300, 366, 400, 500 Nautical Miles (Circular)

Orbit Inclinations: 97.55, 98.09, 98.30, 99.10 Degrees (South-Heading)

Descending Node Times of Day: 0930, 1030, 1200, 1330 Hours

Times of Year: Vernal Equinox, Winter Solstice, Summer Solstice

In order to provide flux data for the EOS/Titan and for the two EOS/Delta spacecraft configurations considered, a generalized 30 surface flux model was developed. A two surface flux model was used for the solar array.

Figure 1.2-1 shows the subsystem module locations assumed for the EOS/Titan and for the two EOS/Delta configurations (called Delta 1 and Delta 2) and identifies each module on each spacecraft with a corresponding surface on the flux model. Figure 1.2-1 also summarizes the absorbed external heat fluxes for each module location for all 14 computer runs and identifies the orbit condition at which maximum and minimum fluxes occur.

Solar array emission to the spacecraft was computed at the conditions of maximum and minimum external absorbed flux for the C&DH subsystem location on the Titan and Delta 1 configurations and for the ACS subsystem location on the Delta 2 configuration. Solar array orbit average temperatures were determined for the appropriate spacecraft worst case flux condition and array emitted fluxes established. Since the solar array rotates with respect to the spacecraft as the EOS travels around the earth, it was necessary to determine the instantaneous configuration factor between spacecraft and solar array at a number of points in the orbit in order to obtain an average value. Descending node time of day (DNTD) of 0930 and 1200 define the extremes of solar array tip angle which were considered. Thus configuration factors were determined for both Titan andDelta configurations, for both 0930 and 1200 DNTD's and at four positions in each orbit to establish the appropriate configuration factor between spacecraft and solar array.

Table 1.2-3 lists the incident orbital average flux components that comprise the maximum and minimum fluxes for all subsystem modules on the Titan and Delta spacecraft. These fluxes include the direct solar, albedo, earth IR and solar array incident heat fluxes which can be used for parametric design studies for the subsystem modules. It should be noted that the worst case flux is based on the absorbed solar, albedo and earth IR fluxes for a skin with an $\alpha_s = .15$ and $\epsilon_{TH} = .75$. A different set of skin properties or the solar array flux could change the condition at which the worst case flux occurs. The information contained in Table 1.2-3 is deemed sufficient for the purpose of subsystem location trade studies and for establishing a baseline thermal control system for the subsystem modules.

C-2

1.2.4 Structure Thermal Analysis

The effects of thermal control heater power on average structure temperature has been determined for the separate structural elements that comprise the EOS structure. These elements include the subsystem module structure, instrument support structure, orbit adjust structure and the transition ring. An insulated Delta 2910 spacecraft was the assumed configuration for the analysis. This configuration was made for an MSS/HRPI instrument compliment. Although a specific configuration was evaluated, the approach and results should be indicative for all configurations which may result.

Assumptions

The following assumptions apply in general for all structure elements:

- o insulation - multilayer, with effective emittence range between 0.2 to 0.5
- o outer layer properties - $\alpha = .45$, $\epsilon = .60$
- o orbit - 366 nautical mile, 0930 DNTD, summer solstice

The assumed thermal design of all structural elements is to provide multilayer insulation blankets on the exterior surfaces of all decks, webs, struts, etc. thus eliminating the need for large areas of skins in such locations as the instrument structure. The outer layer of insulation consists of a soft shield (aluminized Kapton, Kapton side out) which provides a relatively stable thermal coating.

Adjacent structural elements are assumed thermally isolated from each other and from the particular component supported., i.e., instruments are conductively decoupled from the instrument structure, subsystem modules are decoupled from the module structure and RCS engines are decoupled from the orbit adjust structure.

Figures 1.2-2, 1.2-3 and 1.2-4 respectively show the heater power versus structure temperature for the instrument support structure (MSS and HRPI instruments), the module structure and for the orbit adjust structure. Sketches included in each figure indicate the configuration analyzed, the overall dimensions and the assumed adiabatic interfaces. Figures 1.2-5 and 1.2-6 give similar information for the assumed transition ring configurations for the Delta and Titan vehicles, respectively. Both baseline and preferred transition ring concepts are shown in each figure. In each preferred concept, all exposed surfaces except those actually making contact are insulated on the exterior. Exposed areas are assumed coated with a thermal finish with an $\alpha/\epsilon \approx 1.0$ but individual α and ϵ adjustable. Effective emittance of the insulation was treated as a parameter.

Structure Heater Power

The relatively benign thermal environment for the EOS structure in the low altitude sun synchronous orbit coupled with the relative high α/ϵ ratio assumed for the structure exterior coating results in adiabatic exterior surface temperatures in the order of -40°F to $+40^\circ$F, which in turn result in warm structure temperatures with only modest amounts of heater power. Structure heater power has been further reduced by minimizing surface areas such as by eliminating instrument closure skins and substituting a strut supported deck. This approach results in significant cost and weight savings based on data presented in Table 1.2-1 for thermal skins. However, with this concept, the instrument package is thermally decoupled from the structure and must provide its own temperature control. Preliminary contact with instrument vendors indicate the insensitivity of instrument temperature to support structure temperature thereby con-

confirming suitability of this design approach. The results of the structure heater power study are summarized in Table 1.2-2, which shows the total heater power required to maintain the structure at 70°F with a very modest blanket effective emittance value of .05, is 66 watts. This heater power can be reduced to 28 watts by obtaining better blanket performance (effective emittance of .02). Providing a colder structure temperature of 40°F (the minimum value required by the RCS propellants) reduces the heater power range to 15-38 watts.

TABLE 1.2-2
SUMMARY OF STRUCTURE HEATER POWER - DELTA CONFIGURATION
(1) HEATER POWER (WATTS)

Structure Component	Structure Temp = 40°F		Structure Temp - 70°F	
	ϵ Eff=.02	ϵ Eff=.05	ϵ Eff=.02	ϵ Eff=.05
Subsystem Module	5	14	11	27
Instrument				
- upper deck	1	3	2	6
- lower beam	6	15	10	23
Orbit Adjust Structure	2	5	4	9
Transition Area(2)	< 1	< 1	1	1
TOTAL	15	38	28	66

(1) Based on alum. Kapton outer shield (α = .45, ϵ = .60)
(2) Delta Preferred - six fittings

1.2.5 Future Efforts

During the next study phase an integrated structure comprehensive thermal model will be developed to refine the analysis (i.e., thermal gradients) and verify the the approaches. The impact of other missions will also be considered.

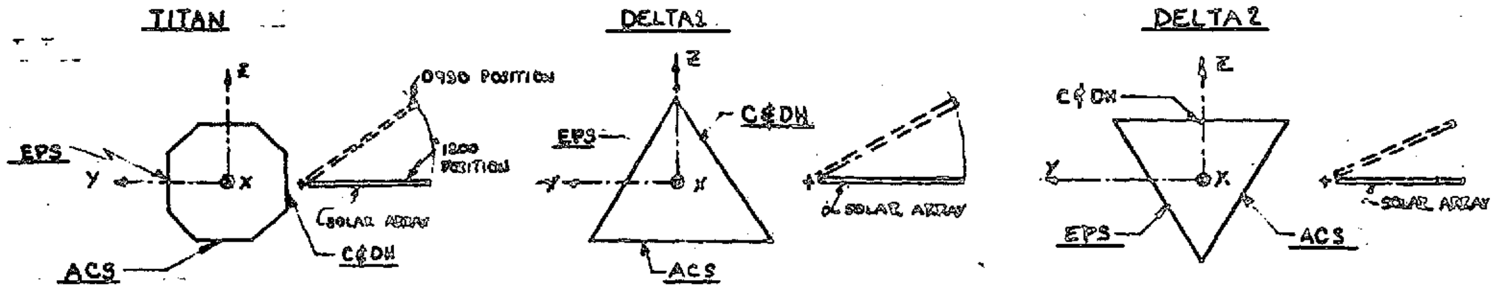
TITLE	TRADE STUDY REPORT NO.
	WBS NUMBER

TABLE 1.2-3

EOS WORST CASE INCIDENT FLUX SUMMARY

SURFACE NO.	INCIDENT FLUX MAX/MIN (BTU/HR FT^2)					
	SOLAR(S)	ALBEDO(A)	EARTH IR(E)	PADDLE(P)	S+A	E+P
1	10.8/17.1	34.1/30.8	62.5/56.7	0/0	44.9/47.9	62.5/56.7
5	141.5/109.4	0/0	0/0	0/0	141.3/109.4	0/0
10	14.9/0	9.3/5.4	17.6/15.1	0/0	24.2/5.4	17.6/15.1
13	184.1/0	9.0/8.1	17.6/15.1	70.5/68.5	193.1/8.1	88.1/83.6
27	5.4/0	20.1/12.3	36.9/31.9	0/0	25.5/12.3	36.9/31.9
28	76.2/0	2.6/1.2	4.9/3.6	0/0	78.8/1.2	4.9/3.6
29	206.4/68.0	2.6/1.9	4.9/3.6	6.9/7.9	209.0/69.9	11.8/11.5
30	112.5/8.6	17.2/17.3	35.1/31.9	18.5/7.9	129.7/25.4	53.6/39.8

S/C CONFIGURATIONS



FLUX MODEL SURFACE ID

	FLUX MODEL SURFACE NO.		
MODULE	TITAN	DELTA 1	DELTA 2
EPS	10	27	28
C&DH	13	30	1
ACS	5	5	29

EXTERNAL ABSORBED HEAT FLUX (1) (BTU/HR FT²)

FLUX RUN NO.	DESCENDING NODE TIME OF DAY	SEASON	ALT. (NM)	SURFACE 1	SURF 5	SURF 10	SURF 13 (2)	SURF 27	SURF 28	SURF 29 (2)	SURF 30 (2)
1	0930	Ver. Eq.	366	52.0	16.4	14.2	40.9	28.4	3.9	33.8	44.8
2	1030	"	"	52.2	19.0	14.4	30.6	28.9	5.4	26.2	36.0
3	1200	"	"	52.3	20.4	14.6	14.6	30.2	14.2	14.2	30.2
4(3)	1330	"	"	52.2	19.0	30.6	14.4	36.0	26.2	5.4	28.9
5	0930	"	300	53.2	16.4	15.4	41.5	29.9	4.5	34.1	45.6
6	1200	"	"	53.6	20.4	15.8	15.8	31.5	14.9	14.9	31.5
7	0930	"	400	51.4	16.4	13.6	40.7	27.7	3.6	33.6	44.5
8	1200	"	"	51.6	20.4	14.0	14.0	29.5	13.9	13.9	29.5
9	0930	"	500	49.7	16.5	12.2	39.9	25.8	2.9	33.2	43.6
10	1200	"	"	49.7	20.4	12.6	12.6	27.8	13.2	13.2	27.9
11	0930	Win. Sols.	366	52.2	16.9	14.2	42.2	28.5	3.9	35.0	45.8
12	1200	"	"	52.7	21.2	14.7	17.1	29.9	13.0	16.3	30.9
13	0930	Sum. Sols.	"	51.8	17.2	14.3	35.0	28.6	3.9	29.3	39.2
14	1200	"	"	52.2	19.8	16.8	14.6	30.6	15.4	12.4	29.7

(1) Orbital Avg solar, albedo and earth emission absorbed by surface with $\alpha_S = .15$ and $\epsilon_H = .75$

(2) Additional Solar Array Flux required for this surface

(3) 1330 orbit hour angle not considered for design

▭ Max Abs. Ext Flux

◯ Min Abs Ext Flux

1.2-12

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Fig. 1.2-1 EOS Subsystem Module External Abs. Heat Flux

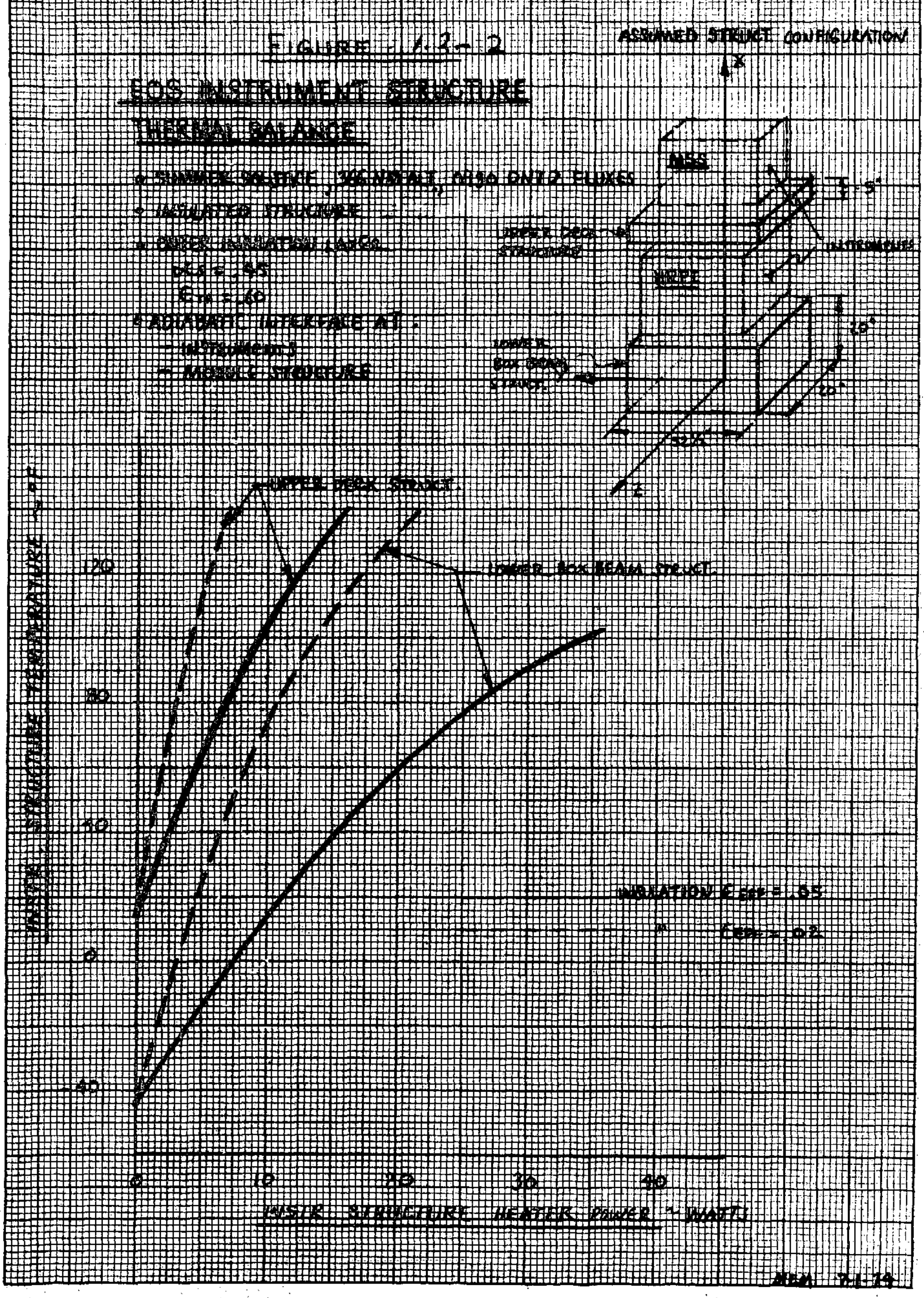


FIGURE 1.2-2
EOS INSTRUMENT STRUCTURE
THERMAL BALANCE
• INSULATED STRUCTURE
• ADIABATIC INTERFACE AT:
— INSTRUMENTS
— MODULE STRUCTURE
ASSUMED STRUCT CONFIGURATION
X
Z
5"
20"
20"
32"
UPPER DECK STRUCT.
LOWER BOX BEAM STRUCT.
INSULATION ε EFF = .05
" ε EFF = .02
120
80
40
0
-40
0
10
20
30
40
INSTR. STRUCTURE HEATER POWER ~ WATTS

FIGURE 1.2-3

EOS MODULE STRUCTURE THERMAL BALANCE

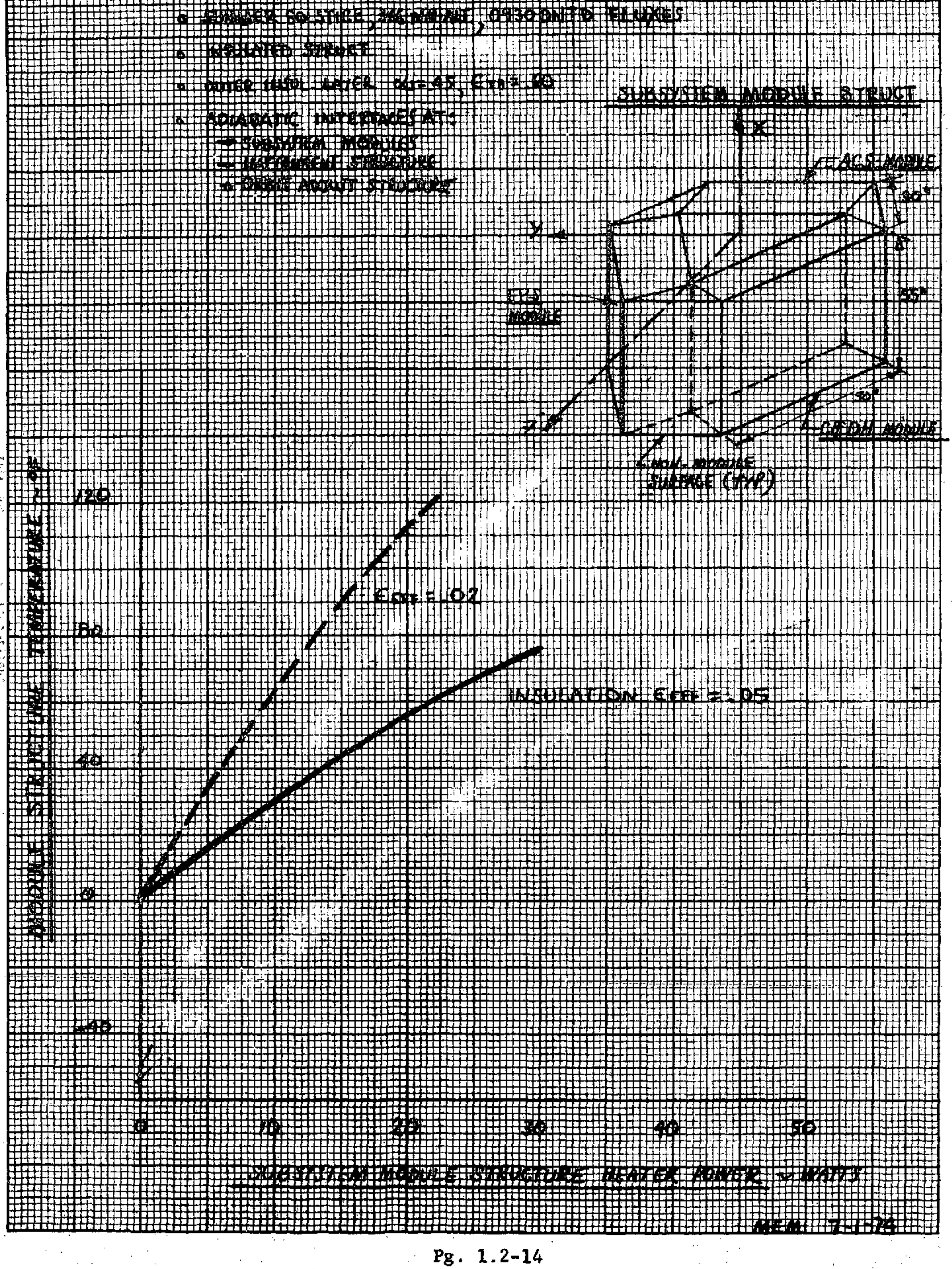


FIGURE 1.2-4

EOS ORBIT ADJUST STRUCTURE THERMAL BALANCE

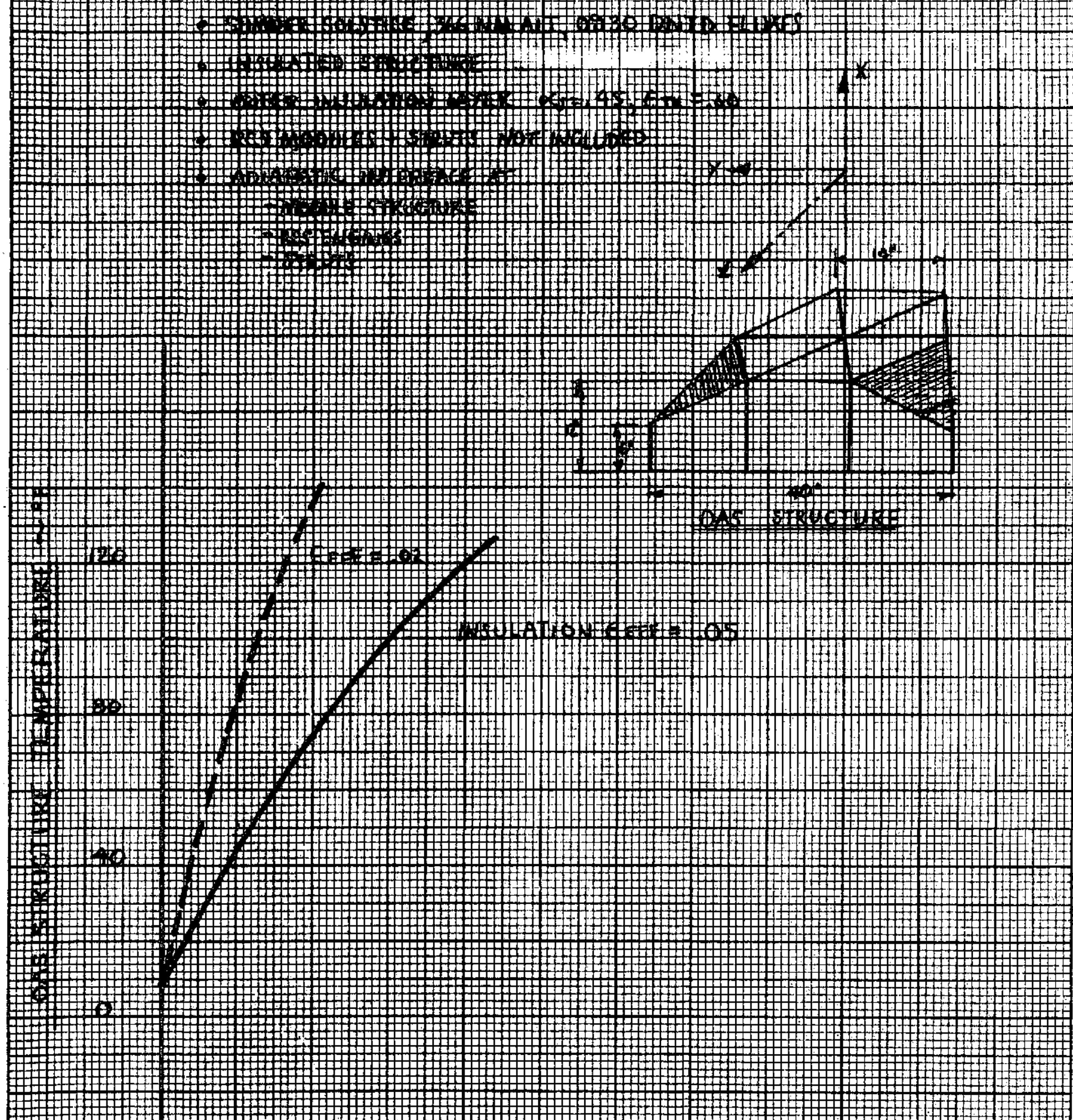


FIGURE 1.2-5

EOS TRANSITION RING THERMAL BALANCE - DELTA CONFIG

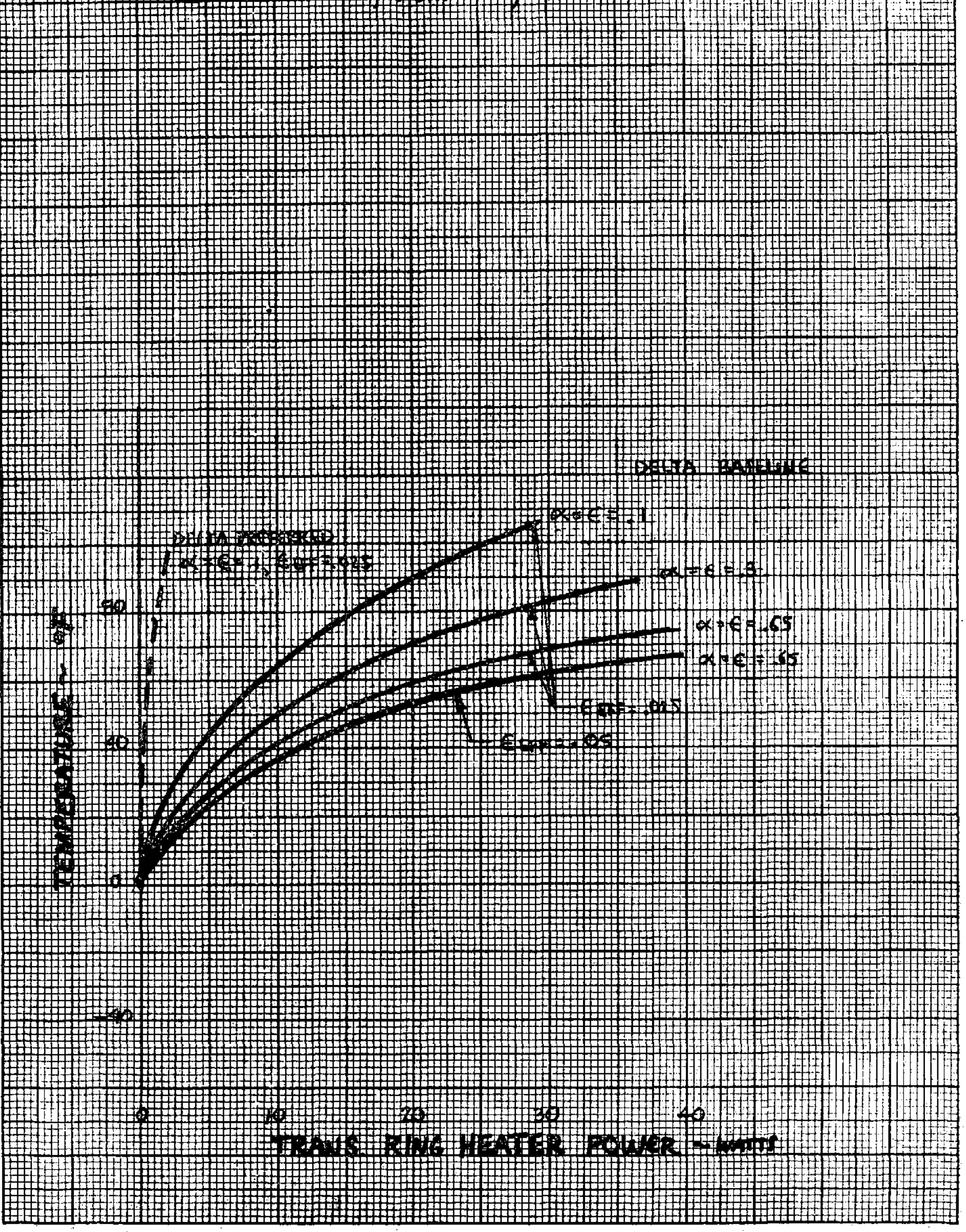
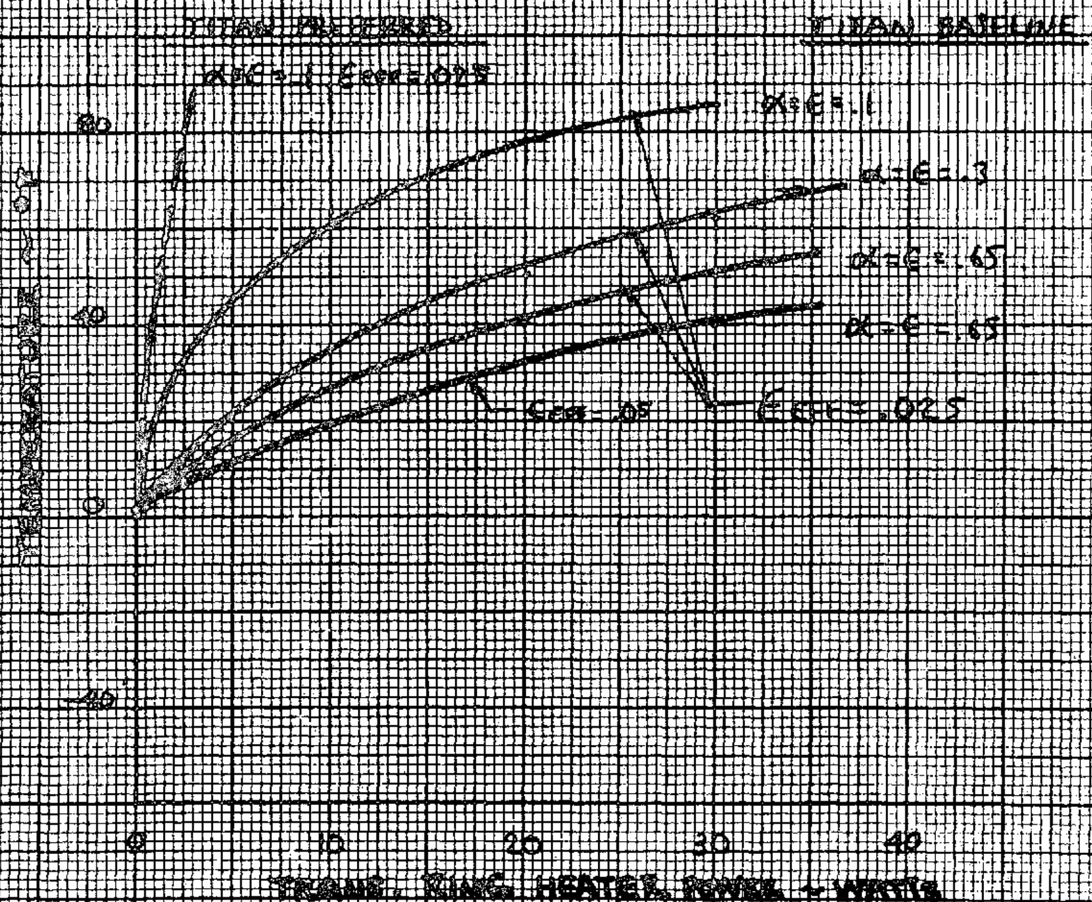


FIGURE 1.2-6

EOS TRANSITION RING THERMAL BALANCE - TITAN CONFIG.

@ SUMMER SOLSTICE, 366NM ALT., 0930 DNTD



1.3.1 Attitude Control

1.3.1.1 Requirements

The basic requirements for the ACS are summarized in Table 1.3.1-1 The range of external disturbance torques results in the need for different sizes of reaction wheels and magnetic torquer bars. The range of missions - from Earth pointing to Stellar and from low to geosynchronous altitude - results in the need for the update sensors to be capable of operating at low altitude orbit rate, geosynchronous altitude orbit rate, and at zero rate. Modal requirements are repeated and expanded in Section 1.3.1.6, under modal operations.

1.3.1.2 Alternative ACS Configurations

Based on the ACS requirements as given in Table 1.3.1-1, three candidate ACS configurations were established: 1--meets requirements lower than baseline (0.05 deg attitude accuracy and 3×10^{-4} deg/sec angular rate stability), 2--meets baseline requirements (0.01 deg attitude accuracy and 10^{-6} deg/sec angular rate stability), and 3--meets requirements higher than baseline (0.002 deg attitude accuracy and 0.2×10^{-6} deg/sec angular rate stability). These ACS configurations are summarized in terms of components, cost, weight, and performance in Tables 1.3.1-2, 3, and 4. The components that change with configuration are the sensors (rate gyros, startrackers, and earth sensor) and the associated software in the C&DH OBC (which is not included in the component listings). Each configuration has 3 different sizes of wheels and bars: size 1 for spacecraft up to approximately 8500 lb, size 2 for spacecraft between 8500 and approximately 17,000 lb, and size 3 for spacecraft between 17000 and 25000 lb. The size 1 magnetic torquer bars are used with the size 1 reaction wheels; similarly, for sizes 2 and 3. Whenever possible, the components selected are space qualified, and when not space qualified are presently in development.

The capability to handle solar and stellar missions in addition to the earth pointing missions is present in ACS Configurations 2 and 3 but not in 1.

The capability of the ACS Configuration 1 exceeds that of the ERTS-A. This system is the least costly, complex, and versatile. ACS Configuration 2 is the baseline design, in which gyro control is normally maintained, with updates using a fixed-head star tracker. Extensive use is made of the C&DH OBC. This system is of medium cost, complexity, and versatility. The range of missions capable of being satisfied include earth-pointing, stellar, and solar. In ACS Configuration 3, a gimbaled star tracker having high resolution and accuracy is used to achieve the highest accuracy obtainable within the current state-of-the-art. Candidate gimbaled star trackers are the TRW proposed (unqualified) of PADS, the Bendix Skylab, and the Kollsman Instrument (OAO-type). Both the Bendix and Kollsman gimbaled star trackers are space qualified but require modifications to incorporate high-resolution angle resolvers.

The three ACS configurations are compared on a cost, weight, and performance basis in Table 1.3.1-5. The costs and weights vary with the use of different-size reaction wheels and magtorquer bars. The weights for all sizes remain below the 600 lb prescribed in the requirements of Table 1.3.1-1. The recurring cost varies from 0.638 $M (ACS Config 1, size 1) to 1.370 $M (ACS Config 3, size 3).

1.3.1.3 Selected Configuration

The selected configuration (Table 1.3.1-3) is adequate for the earth-pointing (low and geosychronous orbit altitudes), stellar/inertial, and solar missions in most respects. Sensors are available in flight proven design with adequate accuracy and sensitivity. Reaction wheels providing up to 8.5 ft-lb-sec and 6 in-oz torque have also been flight qualified. These wheels can be easily qualified to 20 ft-lb-sec and 15 in-oz with minimal development cost/risk. Larger wheels capable of 50-100 ft-lb-sec and 25-50 in-oz are under development and would be available for those missions requiring them. The concept of providing the ACS control algorithms as a mission peculiar software program to be processed in the OBC is viable.

Use of a single FHT with magnitude threshold at approximately 3.3 to provide sufficiently frequent updates to keep gyro drift less than 2.2 sec/20 min is feasible. However, at synchronous orbit with the same magnitude threshold, a single FHT may not have sufficient guide stars available to permit an update every 20 minutes. Consideration should be made to (1) increasing FHT sensitivity, (2) using additional FHT (s) with different view angles, and (3) using a gimbaled star tracker to provide more continuous coverage. The latter approach is costly.

A description of each component in the selected ACS configuration is given in the following paragraphs. A block diagram of the system is given in Fig. 1.3.1-1.

(1) Coarse Sun Sensor

Sun Sensor information is required along two axes, namely the axes which lie in the plane of the solar array. During acquisition, the sunsensor provides 4π steradian coverage so that the sun can be acquired from any orientation of the vehicle. The coarse sunsensor system proposed is the Bendix WASS model #1771858, as described in Figs. 1.3.1 - 2A and 2B. The wide angle sun sensor field of view is considerably in excess of a hemisphere so that two such units would suffice for acquisition over the full sphere without precise alignment. The unit consists of a basic four-quadrant set of photocells arranged in dual opposition, and a set of peripheral cells for angles far off axis. The output signal is analog. The specifications for the sensor are as follows:

Output	0 to 5 milliamps
Impedance Load (External)	100 ohms
Sensitivity - Null	0.20 ma/deg
Temperature Range	-70°C to +50°C
Weight	2.5 oz.

This component is fully space qualified.

(2) Digital Sun Sensor

The fine sun sensor proposed for EOS is a high-resolution digital solar sensor with an accuracy exceeding 1 minute of arc and a field of view of 32 x 32 degrees. The resolution of this device is 1/256 degrees, or 14 arc seconds. The Digital Sun Sensor is used during initial acquisition, and for gyro update. The basic principle of its operation is shown in Fig. 1.3.1-3. A gray-coded pattern on the bottom of a quartz block screens light passing through a slit on the top of the block to either illuminate or not illuminate each of the photocells. The angle of incidence determines which photocells are illuminated. The photocell outputs are amplified and presence of a "1" or "0" is stored in a register to provide the output for use in the attitude computer as well as in telemetry. The unit is fully space qualified.

(3) Gyro Assembly

The Gyro Assembly functionally outputs three digital 16-bit words proportional to the input inertial rates. The assembly consists of an orthogonal triad of rate-integrating gyros configured for closed-loop (rate) operation. The gyro triad is hard mounted to the vehicle so that the gyro input axes are colinear with the roll, pitch and yaw axes of the spacecraft.

They Gyro Assembly is configured around the Bendix 64 PM RIG single degree-of-freedom gyroscopic sensing unit. The gyroscopic unit incorporates a hydrodynamic spin motor (wheel) within a cylindrical float which is supported (suspended) by a self-contained, hydrostatic liquid bearing.

Essentially, the EOS unit is a derivation of the 25IRIG. The angular momentum of the unit is changed by raising the wheel speed. The basic techniques successfully used to produce the 25IRIG have been applied to the development and production of the proposed EOS Gyro, including its Hydrodynamic Wheel Gas Bearing, the Hydrostatic Gimbal Liquid Bearing, the Torquer, the Pickoff, and the Fluid.

The unit will require a qualification program to satisfy EOS needs. The significant performance features of the gyro assembly are given in Table 1.3.1-6.

(4) Fixed Head Tracker

The Fixed Head Tracker (FHT) provides attitude sensing information for attitude determination and gyro assembly update. The FHT is built by ITT and is based on their existing Fine Guidance Error System which has successfully flown on over seventeen Aerobee rocket flights, and is presently being modified for the ELMS program. The FHT consists of the following subassemblies:

- Star Aspect Sensor (SAS)
- Power Supply (PS)
- Bright Object Sensor (BOS)
- Earth Albedo/Sun Shade (EA/SS)

The FHT provides two-axis star position signals for identifiable stars passing the through the field of view. The FHT also provides data such as relative star brightness and sensor temperature to enhance the determination of space vehicle attitude. A functional block diagram of the FHT is presented in Fig. 1.3.1-4.

The FHT has two modes of operation, a TV-type search scan and a star angle detection track mode. In the search mode, the search track generator produces a step scan in which the output of the image dissector tube is monitored for presence of a target star. The brightness range of the stars which may be tracked is determined by ground command, in that the tracker's sensitivity can be changed in one-star-magnitude increments over four steps. Upon the detection of a star brighter than the minimum set threshold level, the tracker automatically switches from search to track mode. A cross-scan sweep is employed which in turn produces a pulse-width-modulated video signal. Demodulation circuits produce a signal which maintains the cross scan centered on the target star. The angle between the electrical null and the target star is generated by sampling the image dissectors' deflection coil currents. As movement between the target star and the spacecraft occurs, the FHT automatically maintains the center of the cross scan on the target star as long as it remains in the 8° field of view. Upon loss of the target star, the FHT will automatically switch to the search mode until the next suitable star is acquired. The FHT can also be commanded to switch from track to search mode by ground control.

The shutter is built into the EA/SS which is operated by the BOS. A power-reduction circuit is included to reduce the power dissipation of the FHT when the shutter is operated. This reduction will be in two areas. The SAS will be shut off whenever the shutter is closed, and the voltage to the shutter solenoid will be reduced after the shutter is closed. The BOS will be operational shortly after the shroud is jettisoned in order to protect the FHT. The BOS will be capable of being powered on and off via ground control and the shutter is designed such that a failure in operation will fail open.

The present SAS is a pressurized unit which eliminates the problems associated with high voltage breakdown and outgassing from potting and encapsulating material. The image dissector tube being used is an electrostatic focused FW 143 with an S-20 photo cathode. The unit will be qualified in the ELMS program.

The startracker will be mounted inside the ACS module with the startracker sunshade entirely inside the module and flush to the top panel. The tracker will be tilted 45° from the spacecraft -Z axis (anti-earth) to the right in the ZY plane toward the +Y axis. For a 9:30 AM to 12 Noon orbit, the sun will be 45° (during earth-pointing) at its closest (in the case of a 12 Noon orbit). For afternoon orbits, the spacecraft is yawed 180 degrees, and the sun would be no closer than 45° from the startracker null axis in this case also (12 Noon orbit).

The significant performance features of the FHT are as follows:

- Star Threshold

The SRA has four commandable threshold levels. The FHT operates normally when the target stars are within the minimum and maximum range described as follows:

Minimum star; +5.30 visual magnitude, GO V or bluer spectral class

Maximum star; +2.0 visual magnitude, GO V or redder spectral class

The minimum star detectability will not vary by more than ± 0.30 magnitudes across the entire search field for each of the commandable thresholds listed below. The minimum intensity or star signal which can be tracked shall be adjustable on-orbit to 2, 3, 4, and 5 magnitude levels.

- Star Motion

A target star moving through a field of view at angular rates as large as 0.6 degrees per second is acquired and tracked for all star brightness levels indicated above.

- Search Period

The maximum time required to search the total field of view is less than 2 seconds. The probability of successful acquisition of an acceptable target star is greater than .90 for any one search period.

- Noise Equivalent Angle (NEA) -

The short term deviation of the output star position signals, while observing the minimum detectable target star which is stable in both angular positions in the field of view and in irradiance, has an rms value of 22 arc seconds or less when the bandwidth is 4.0 hz.

- Calibrated Accuracy

The repeatability of variations in gain, linearity, and distortions in response to all environmental and functional parameters (such as temperature, magnetic field, star intensity, and position in the field of view) are such that the true position of the star with respect to electrical zero can be determined with a one standard deviation error no greater than 20 arc seconds, accomplished by applying correction factors to the output signals. The factors for correction of the effects of temperature, magnetic field, star intensity and position in the field of view are measured on each FHT during manufacture. Thermal hysteresis effects, if present, are correctable with an additional correction factor, but all efforts are made to keep them to a minimum. The Star sensor accuracy as a function of various parameters is given in Table 1.3.1-7.

(5) Electronics Assembly

The EA proposed by Ithaco, Inc. contains the major portion of the control system electronics. It has within one envelope the following operations:

- Signal conditioning
- Analog Processor
- Wheel Drivers
- Magnetic torquer drivers
- Jet drivers
- Magnetometer electronics

The EA will be made in a modular arrangement of electronic cards in metal frames with inter-connections made by a wiring harness. The unit requires a qualification program to satisfy EOS needs.

6) Reaction Wheels

The Bendix N&C division has been selected for the proposed reaction wheels. The size 1 wheel will be similar to the one flown on OAO, which is space qualified.

Reaction Wheel Design

The RW is a hermetically sealed unit filled with an inert gas mixture of 98% helium and 2% oxygen. This combination offers a low density/low windage drag advantage as well as an accurate leak rate measurement. RW leak rate measurements are made by Bendix during component testing.

The RW bearings are single-row, double-shielded, deep-groove type, with a two-piece ribbon cage. The ball and raw material are AISI 52100 CEVM chrome alloy steel. The bearings shall be R-10 size, tolerance class ABEC-7, lubricated with 25 milligrams of Type SRG-40 mineral oil. Because the reaction wheel runs through zero speed, and, therefore, cannot generate a hydrodynamic film for a portion of its operating speed range, the bearings will be lubricated with excess oil (in this case 25 mg). This provides additional assurance that the wheel will operate satisfactorily in the region of lower film thickness.

The configuration incorporates labyrinth seals adjacent to each bearing to control the rate of lubricant evaporation in the event of failure of the housing as a sealed enclosure. To supplement bearing lubricant, sintered nylon oil reservoirs, impregnated with the bearing lubricant, are positioned adjacent to each bearing. Sacrificial evaporation of the lubricant in these reservoirs maintains the vapor pressure of the lubricant in the event of seal failure.

To monitor temperature of the RW, a thermistor is mounted adjacent to the bearing on the motor stator side. This location, due to stator generated heat, is the hotter of the two bearing locations.

(7) Magnetometer -

The magnetometer operates on the flux gate principle. The probe is excited with a 2000 Hertz signal. The probe output contains even harmonics of 2 KHz whose amplitude is proportional to the vector component of the magnetic field aligned along the probe axis. The probe contains a very thin sliver of magnetic material which is the core of a transformer. One winding of the transformer is driven by the 2 KHz sine wave. The presence of earth's steady dc field saturates the thin sliver of magnetic material, and the resulting ac magnetic field, which is sensed by a second winding on the transformer, is distorted and contains a large second harmonic component. The phase of this component in relation to the 2 KHz drive signal is a function of the direction of the earth's steady field. A block diagram of the magnetometer is shown in Fig. 1.3.1-5. This unit is space qualified.

1.3.1.4 Solar Array Direction Reversal

Assuming the use of a flex-lead rather than a slip-ring type of solar array drive unit, requirements are imposed on the solar array drive unit during its negative angular velocity phase which occurs in the dark portion of the orbit. The requirements are as follows:

(1) solar array drive torque in excess of friction $\leq$ 0.005 ft-lb (or 1 in-oz.)

(2) magnitude of the solar array angular velocity relative to the spacecraft $\leq$ magnitude of $\left[(-0.26(200/I)+0.06)\right]$ deg / sec, where I $\hat{=}$ inertia of solar array about solar array drive unit axis.

These requirements ensure (1) that the torques applied to the spacecraft by the solar array drive unit do not exceed the torque capability of the lowest-size reactionwheel (2 in-oz), (2) that the momentum change required in reversing the direction of the solar array is within the momentum capability of the lowest-size reactionwheel (2 ft-lb-sec), (3) that the time required for the solar array to completely return to its position at sunup is less than the time available (approximately 33 minutes).

The operations can be performed as follows. At sunup, the array is driven at a positive rate from a position in which the array is normal to the sunline. The positive rate remains fixed at approximately 0.06 deg/sec with corrections as needed using an array mounted solar sensor. At sundown, the solar array is driven at constant negative rate of approximately 0.18 deg/sec, so that when sunup occurs again the solar array is approximately at the desired orientation.

From the ACS standpoint, it is desirable that the solar array be driven continuously in one direction and as smoothly as possible (not in stepper motor fashion), so that the spacecraft is not perturbed.

1.3.1-5 Candidate Components

The basis for the selection of components in Sections 1.3.1-2 and 3 is the set of components listed in Table 1.3.1-8. In this table, the sellers of a particular type of component are listed, along with the model number, qualification status, range of operation, weight, power and cost.

1.3.1-6 Modal Operations

The equipment and sequence of operations in each mode is a function of the requirements of each mode. Modal operations are described in Table 1.3.1-9. including a listing of the ACS sensors and actuators used in each mode.

Table 1.3.1-1 Basic Requirements For The ACS

ITEM	REQUIREMENT
Missions	Earth, stellar, solar
Mission lifetime Altitudes Spacecraft weight Spacecraft inertias Spacecraft external disturbance torque	2 years operations plus 3 yrs survival 300 to 900 nmi & geosynchronous 2500 to 25000 lbs 500 to 100,000 slug-ft^2 300 to 900 nmi: cyclic peak $\geqslant 2\times10^{-4}$ and $\leqslant 0.2$ ft-lb average $\geqslant 10^{-5}$ and $\leqslant 0.1$ ft-lb Geosynchronous altitude: 10% of the values given for 300 to 900 nmi
ACS Modes	Acquisition, slew (single-axis), Inertial Attitude Hold, Earth-oriented mission, Stellar Mission, Solar Mission, Survival, Orbit Trim, and Orbit Adjust. Also may require SRM burn.
Acquisition Mode	Separation rates $\leqslant$ 1 deg/sec Final attitude $\leqslant$ 2 deg Final angular rates $\leqslant \pm$ 0.03 deg/sec
Slew (single-axis) Mode	Slew angle $\leqslant$ 90 deg. Accumulated error $\leqslant \pm$ 0.03 deg. Rate of slew $\geqslant$ 2 deg/min
Inertial Attitude Hold Mode	Drift before in-orbit calibration $< \pm$ 0.03 deg/hr Drift after in-orbit calibration $< \pm$ 0.003 deg/hr
Earth-oriented Mission Mode	Point yaw axis to earth centroid Orbits: (1) sun-synchronous (9:30 am - 12 Noon) circular 300-900 nmi (2) geosynchronous Pointing accuracy/axis < 0.01 deg Pointing stability/axis: (1) average rate deviation over 30 min $< \pm 10^{-6}$ deg/sec (2) attitude jitter relative to average baseline: up to 30 sec $< \pm$ 0.0003 deg up to 20 min $< \pm$ 0.0006 deg In-orbit calibration acceptable

Table 1.3.1-1 Basic Requirements For The ACS

(continuation)

ITEM	REQUIREMENT
Stellar-Mission Mode	Time interval $\leq$ 1 hour Pointing accuracy/axis $\leq$ 0.01 deg Pointing stability/axis: (1) average rate duration over 30 min $< \pm 10^{-6}$ deg/sec (2) attitude jitter relative to average baseline $< \pm 0.0006$ deg With perfect instrument error signals: (1) Pointing accuracy/axis $< \pm 3X10^{-6}$ deg (2) Pointing stability/axis: attitude jitter relative to average baseline $< \pm 10^{-7}$ deg
Survival Mode	Solar array $\perp$ sunline $< \pm 7$ deg Angular rate/axis $<$ 0.05 deg/sec Time: continuous. Reliability: 95% Support Shuttle resupply and retrieval
ACS Interface	Instrument: have capability for using Instrument pointing error signals Pneumatics: send on-signals to jets C&DH OBC: send signals to C&DH OBC & receive signals from OBC
ACS Cutoff Frequency	Approximately 0.1 hz
Reaction Wheels	Number of selectable units $\leq$ 4 Interchangeable electrically & physically
Magnetic Torquers	Magnetic field at external envelope of ACS module $\leq$ 0.1 gauss
ACS Module	Dimensions 48" x 48" x 18" Weight $\leq$ 600 lbs Power $<$ 150 watts
Mass Expulsion	Two torque levels for on-orbit operation: (1) high torque for initial stabilization, orbit adjust, & backup for reactionwheel in survival mode (2) low torque for backup momentum unloading of reactionwheel and possibly a backup for reactionwheel in pointing mode In case SRMs are used, a third level of jet torque is generally required.

Table 1.3.1-2 ACS Configuration 1

COMPONENT	NUMBER/ SPACECRAFT	COST, $K NON-RECUR	COST, $K EACH RECUR	WEIGHT EACH LB.	PERFORMANCE
Coarse Sunsensor (Bendix)	2	5	2	0.156	FOV ± 90°, 2 axes
Digital Sunsensor (Adcole)	1	10	42	5	FOV ± 32°, Accuracy 1′, LSB 14 sec
Rate Gyro (Bendix)	1	10	40	7.25	< 1°/hr
Earth Sensor (Quantic)	1	100	125	45	0.02 deg
* Magnetometer (Schoenstedt)	1			6.5	± 1.0 gauss range
* Electronic Assy (Ithaco)	1			13	
Multiplexer (Hughes)	2	0	62	0.5	
Decoder (Hughes)	2	0	10	1.0	
Reactionwheels, size 1 (Bendix)	3	10	30	11.3	h = 2 ft-lb-sec, T=2 in-oz
" " 2 "	3	10	40	20	h = 8.5 ft-lb-sec, T = 7.5 in-oz
" " 3 "	3	100	60	22	h = 25 " " " T = 25 in-oz
* Magtorquer bars, size 1 (Ithaco)	3			10.2	m = 45,000 pole-cm
" " 2 "	3			50	m = 450,000 "
" " 3 "	3			112	m = 4,500,000 "
* Set: 1 Magnetometer, 1 Elec. Assy, 3 Magtorquer bars (size 1)		370	193		
Set: 1 Magnetometer, 1 Elec. Assy, 3 Magtorquer bars (size 2)		422	229		
Set: 1 Magnetometer, 1 Elec. Assy, 3 Magtorquer bars (size 3)		474 (1)	265 (1)		

Notes: (1) Bars cannot fit in ACS module. Estimates provided for comparison purposes only.

Table 1.3.1-3 ACS Configuration 2

COMPONENTS	NO/SC	COST (EA) NON-RECUR	COST (EA) RECUR	WEIGHT (EA)	PERFORMANCE
Coarse Sunsensor (Bendix)	2	5	2	.156	FOV ± 90 deg, 2 axes
Digital Sunsensor (Adcole)	1	10	42	5	LSB 14 sec; accuracy 1 min; FOV ± 32°
Rate Gyro Assy (Bendix)	1	650	235	15.0	.003°/hr (IOC)
Fixed Head Startracker (ITT)	1	40	43	17.0	FOV 8 deg circular Accuracy 20 sec (compensated)
* Magnetometer (Schoenstedt)	1			6.5	Range ± 1 gauss
* Electronics Assy (Ithaco)	1			13	
Multiplexer (Hughes)	2	-	62	0.5	
Decoder (Hughes)	2	-	10	1.0	
Reactionwheel (Bendix, size 1)	3	10	30	11.3	H= 2 ft-lb-sec, T=2 in-oz
Reactionwheel (" , size 2)	3	10	40	20	H = 8.5 ft-lb-sec, T=7.5 in-oz
Reactionwheel (" , size 3)	3	100	60	22	H= 25 ft-lb-sec, T=25 in-oz
* Torquer Bar (Ithaco, size 1)	3			10.2	M= 45,000 pole-cm
Torquer Bar (" , size 2)	3			50	M =450,000 pole-cm
Torquer Bar (" , size 3)	3			112	M =4,500,000 pole-cm
* Set: 1 Magnetometer, 1 Electronics Assy, 3 Magtorquers (size 1)		370	193		
Set: 1 Magnetometer, 1 Electronics Assy, 3 Magtorquers (size 2)		422	229		
Set: 1 Magnetometer 1 Electronics Assy, 3 Magtorquers (size 3)		474	265		

Table 1.3.1-4 ACS Configuration 3

COMPONENTS	NO/SC	COST		WEIGHT	PERFORMANCE
		NON-RECUR	RECUR		
Coarse Sunsensor (Bendix)	2	5	2	.156	FOV ± 90°, 2 axes
Digital Sunsensor (Adcole)	1	10	42	5	LSB 14 sec, Accuracy 1 min, FOV±32°
Rate Gyro Assy (Bendix)	1	650	235	15	Random drift 0.003 °/hr (TOC)
Gimbaled Startracker	1	500	500	50	Gimbal travel ±45° (2 axes) FOV ± 30 min Accuracy 5 sec
Magnetometer (Schoenstedt)	1			6.5	Range ±1.0 gauss
Electronics Assy (Ithaco)	1			13	
Multiplexer (Hughes)	2		62	0.5	
Decoder (Hughes)	2		10	1.0	
Reaction Wheel (Bendix, size 1)	3	10	30	11.3	H = 2 ft lb sec. T= 2 in oz
Reaction Wheel (Bendix, size 2)	3	10	40	20	H = 8.5 ft lb sec. T= 7.5 in oz
Reaction Wheel (Bendix, size 3)	3	100	60	22	H = 25 ft lb sec. T= 25 in oz
Torquer Bars (Ithaco, size 1)	3			10.2	M= 45,000 pole-cm
Torquer Bars (Ithaco, size 2)	3			50	M=450,000 pole-cm
Torquer Bars (Ithaco, size 3)	3			112	M= 4,520,000 pole-cm
Set: 1 Magnetometer, 1 Elect 3 Magtorquers (size 1)		370	193		
Set: 1 Magnetometer, 1 Electronics Assy, 3 Magtorquers (size 2)		422	229		
Set: 1 Magnetometer, 1 Electronics Assy. 3 Magtorquers (size 3)		474	265		

Table1.3.1-5 Comparison of ACS Configurations

ACS CONFIGURATION	HARDWARE COST, $M, PER SPACECRAFT		WEIGHT PER SPACECRAFT LB.	SYSTEM PERFORMANCE	COMMENTS
	NONRECUR	RECUR			
1 (low cost)	Size 1: 0.615 Size 2: 0.667 Size 3: 0.719	Size 1: 0.638 Size 2: 0.704 Size 3: 0.800	Size 1: 144 Size 2: 290 Size 3: 482	0.02 deg 3×10^{-4} deg/sec	Performance is not as good as for Config.'s 2 & 3. Requires more ground processing.
2 (baseline)	Size 1: 1.085 Size 2: 1.137 Size 3: 1.279	Size 1: 0.751 Size 2: 0.817 Size 3: 0.913	Size 1: 124 Size 2: 270 Size 3: 462	0.01 deg 10^{-6} deg/sec	Performance specifications are met. Calibration of star-tracker is done in OBC.
3 (expanded capabilities)	Size 1: 1.545 Size 2: 1.597 Size 3: 1.739	Size 1: 1.208 Size 2: 1.274 Size 3: 1.370	Size 1: 157 Size 2: 303 Size 3: 495	0.002 deg 10^{-6} deg/sec	Pointing performance is improved beyond that of Config. 2.

TABLE 1.3.1-6 PERFORMANCE OF BENDIX 64 PM RIG GYRO

PARAMETER	[illegible] CAPABILITY	BASIS FOR ESTIMATE
BANDWIDTH	≥ [illegible] Hz	64 PM RIG RATE SENSOR EXPERIENCE
GAIN MARGIN	10 db	
PHASE MARGIN	67 DEG	
ATTITUDE MEAS.	–	–
QUANTIZATION	0.1 ARC SEC/COUNT	64 PM RIG RATE SENSOR EXPERIENCE
INPUT RANGE	± 10 DEG/SEC	64 PM RIG RATE SENSOR EXPERIENCE
ATTITUDE SCALE FACTOR	–	–
SCALE FACTOR NON LINEARITY	< 100 PPM ± 0.05–2.8 °/SEC INPUT	64 PM RIG RATE SENSOR EXPERIENCE
SCALE FACTOR STABILITY		
RANDOMNESS	< 10 PPM	64 PM RIG RATE SENSOR EXPERIENCE
1 DAY STABILITY WITH SHUTDOWN	< 100 PPM	64 PM RIG AND 64 PM RIG RATE SENSOR EXPERIENCE
LONG TERM STABILITY	< 1200 PPM	DESIGN ANALYSIS AND 64 PM RIG RATE SENSOR EXPERIENCE
DRIFT CHARACTERISTICS	–	–
G-INSENSITIVE DRIFT		
MAX. VALUE	< 1.0 DEG/HR	64 PM RIG RATE SENSOR EXPERIENCE
SHORT TERM DRIFT	< .001 DEG/HR	64 PM RIG RATE SENSOR EXPERIENCE
DRIFT RATE RAMP	< .0001 DEG/HR^2	64 PM RIG RATE SENSOR EXPERIENCE
PEAK ATTITUDE NOISE	< 3 ARC-SEC	64 PM RIG RATE SENSOR EXPERIENCE AND DESIGN ANALYSIS
SHORT TERM NOISE	< 0.5 ARC-SEC	64 PM RIG RATE SENSOR EXPERIENCE AND DESIGN ANALYSIS
1 DAY STABILITY WITH SHUTDOWN	< 0.05 ARC-SEC	64 PM RIG RATE SENSOR EXPERIENCE
DRIFT RATE (DYNAMIC ENVIRONMENT)	< .003 DEG/HR (MAX)	64 PM RATE SENSOR DESIGN ANALYSIS
G-SENSITIVE DRIFT RATE		
MAX. VALUE	< 1.3 DEG/HR-G	64 PM RIG RATE SENSOR EXPERIENCE
1 DAY STABILITY WITH SHUTDOWN	< 0.4 DEG/HR-G	64 PM RIG EXPERIENCE AND DESIGN ANALYSIS
TEMPERATURE SENSITIVITY	–	–
G-INSENSITIVE DRIFT	.00555 DEG/HR/°F	64 PM RIG EXPERIENCE
SCALE FACTOR	8.3 PPM/°F	64 PM RIG EXPERIENCE
ALIGNMENT	1.5 ARC SEC/DEG F	64 PM RIG EXPERIENCE
VOLTAGE SENSITIVITY	–	–
G-INSENSITIVE DRIFT	< .0005 DEG/HR-VOLT	64 PM RIG EXPERIENCE
SCALE FACTOR	< 5 PPM/VOLT	64 PM RIG EXPERIENCE
MAGNETIC FIELD SENSITIVITY		
G-INSENSITIVE DRIFT	.002 DEG/HR	64 PM RIG EXPERIENCE
SCALE FACTOR	10 PPM	64 PM RIG EXPERIENCE
WHEEL SPIN POWER	< 5 PERCENT	DESIGN ANALYSIS

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Table 1.3.1-7 Star Sensor Accuracy

Error Source	Value, arc-sec (1 σ) Without	With Corrections
Temperature	90	12.0
Mag Field	30	6.0
Field Repeatability	10	10.0
Hysteresis	2	2.0
Log Error - Trk loop Bandwidth 4 hz	2	2
- Error amp	4	4
NEA	22	6.0 +
Total (1 σ) (prelaunch)		18.4 arc-sec ǂ
Null Shift (due to launch)		10.0
Total (1 σ) (on-orbit)		18.4 arc-sec
On-orbit alignment Calibration		4
Telemetry resolution		4
(Total (1 σ)		18.9 arc-sec

* In flight alignment accuracy for roll.

\+ after filtering

ǂ exclusive of null shift

TABLE 1.3.1-8 CANDIDATE COMPONENTS & THEIR ASSOCIATED CHARACTERISTICS

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL STATUS	PROCUREMENT COST, $K		COMMENTS
FUNCTION	SELLER	MODEL	STATUS	FUNCTION/CAPABILITY	WT EA	AVG PWR		RECUR	NON-RECUR	
Coarse Sun Sensor	Bendix	1771858 (SKYLAB)	Exist	4π Steradian Coverage (2 Sensors)	0.156	—	Similarity	4	7	
	Adcole	C-1694 (OAO)	Exist	4π Steradian Coverage (8 Sensors)	0.19	—	Similarity	40	10	
	Adcole	C-1702 ATS	Exist	4π Steradian Coverage (2 Blocks)		—	Similarity	40	10	
Digital Sun Sensor	Adcole	C-1594	Exist	± 32°FOV, 1 min accuracy 14 bits, LSB = 14 sec	5		Similarity	40	10	
Gyro Assy (3 gyros; 1/axis)	Honeywell	GG334	Exist	Direct roll, pitch, and yaw sensing. Jewel dithered suspension. Gas bearing. Short term drift 0.007°/hr	15		Δ Qual			
	Northrup	K76-3C	Exist	Taut wire suspension. Gas bearing. Short term drift 0.002°/hr	—	—	Δ Qual			
	Bendix	64PM-RIG	Exist	Magnetic suspension. Gas bearing	15	25	Δ Qual	205	634	

TABLE 1.3.1-8 (CONTINUATION)

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL STATUS	PROCUREMENT COST, $K		COMMENTS
FUNCTION	SELLER	MODEL	STATUS	FUNCTION/CAPABILITY	WT.	AVG. PWR.		EACH RECUR	NON-RECUR	
Fixed-Head Tracker	Bendix	OAO	Mod	20 arc-sec accy; + 5 mag	16.7	8.7	△ Qual	100	394	
	Ball Bros.	SAS-C	Mod	20 arc-sec accy; 3, 4, 5 mag commandable	13.8	6.0	△ Qual	77	65	
	ITT	ELMS	Exist	20 arc-sec accy; 3, 4, 5	17	9	Similarity	43	83	
Electronics Assy	Ithaco	—	New	Assy consists of 1) signal conditioning 2) analog processor 3) wheel drivers 4) mag torquer driver 5) magnetometer electr. 6) magnetometer (3) 7 Torquer bars (3)	13 6.5 *	15	Required	193.2 229.1	370.4 422.4	*Three sizes are proposed 1) 45 amp-meter² 2) 450 " " 3) 4500 " "
Reaction Wheels (3)	Bendix	1880272 OAO	Exists	Torq. = 2 in-oz H = 2.06 ft-lb-sec NLS = 1200 rpm	10	2.3	Similarity	30	40	
	Bendix	188026 ATS	Exists	Torq. = 20 in-oz H = 8.47 ft-lb-sec NLS = 1500 rpm	19.5	10	Similarity	40	40	
	Bendix		Mod	Torq. = 25 in-oz H = 25 ft-lb-sec NLS = 3000 rpm	22	18	△ Qual	60	100	

TABLE 1.3.1-8(CONTINUATION)

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL STATUS	PROCUREMENT COST, $K		COMMENTS
FUNCTION	SELLER	MODEL	STATUS	FUNCTION/CAPABILITY	WT EA	AVG PWR		RECUR	NON-RECUR	
Gimbaled Startracker	KIC	OAO	Exist	FOV 1' x 1'; Gimbal travel, both, ± 43° Accuracy 5 sec	50	15.4	Similarity			
	Bendix	OAO	Exist (did not fly)	FOV 1' x 1' Gimbal travel, both, ± 43°			Similarity			
	TRW	PADS	Mod	FOV 0.5' x 0.5' Gimbal travel: inner ± 15° outer ± 45°	50	28	△ Qual			
Horizon Sensor	Barnes	13-166	Mod	2-axis earth sensing ± 1°	21.2	20	Similarity	195.5	16.6	
	Quantic		Exist	± 0.02°	45	20	"	125	100	

TABLE 1.3.1-9 MODAL OPERATIONS

PHASE	MODE		ATTITUDE CONTROL REQUIREMENTS	CONTROL SENSORS	CONTROL ACTUATORS	DESCRIPTION OF MODE
EOS Separation	Null separation rates		Separation rates < 1 d/s. Final angular rates $\leq \pm$ 0.03 d/s	R,P,Y rate gyros	R,P,Y 1-lb jets	Following separation from booster, spacecraft rates are nulled.
	Inertial hold		Angular rates $\leq \pm$0.03 d/s	R,P,Y rate gyros	R,P,Y 1-lb jets	
Power up	Coarse Solar point		Align spacecraft -Z axis to sun < 2 d. Final Angular rates $\leq \pm$0.03 d/s	R,P CSS zenith along -Z axis (mounted on ACS Module). R,P,Y rate gyros	R,P,Y 1-lb jets	
	Deploy solar array		Maintain spacecraft -Z axis to sun < 2 d. Maintain angular rates $\leq \pm$0.03 d/s	R,P CSS zenith along -Z axis (mounted on ACS Module) R,P,Y rate gyros	R,P,Y 1-lb jets	
	Orient Solar array normal to sun		Align solar array normal to sun < 7 d. Final angular rates $\leq \pm$0.03 d/s	CSS mounted on solar array. R,P,Y gyros	R,P,Y 1-lb jets	Solar array drive unit is locked.
	Maintain solar array normal to sun		"	"	R,P,Y reactionwheels with magnetic unloading.	"
Checkout						Approximately 1 month. Star trackers are not turned on for a minimum of 3 days (unless star tracker design permits it) to prevent arcing due to outgassing.
	Initial Precision Update	Coarse Solar Point	Same as above for Coarse Solar Point			
		Fine Solar Point	Align spacecraft -Z axis to sun < 0.1 d. Final angular rates $\leq \pm$0.03 d/s	R,P DSS zenith along -Z axis (mounted on ACS Module). R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading	Update attitude computation in OBC. Utilize DSS and magnetometer signals for 3-axis update. (Ephemeris, & model of Earth's magnetic field is used)
		Slew about Z axis (sunline)	Achieve slew rate in $\leq$ 4 m. Slew rate $\geq$ 2 d/m. Reduce to zero rate in $\leq$ 4 m.	R,P,Y rate gyros R,P DSS	R,P,Y 1-lb jets	The slew angle is calculated to arrive at an attitude in which the FHT should see a guide star.
		Inertial Hold	Angular rates $\leq \pm$0.03 d/s	R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading	FHT searches for its star. If found, FHT goes into track mode. Update using DSS &FHT. If star not found, assess, & use alternate procedure.
	Calibrate Gyros	Inertial Hold	"	"	"	Using updates from FHT and DSS, calibrate gyros. Insert rate bias errors in OBC.
	Earth-Point	Slew	Achieve slew rate in $\leq$ 4 m. Slew rate $\geq$ 2 d/m. Reduce to zero rate in $\leq$ 4 m	R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading.	Using sequential slews, slew to attitude which at a later point in the orbit results in an Earth-pointing attitude. Update with FHT during slews.
		Inertial Hold	Angular rates $\leq \pm$0.003 d/s	R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading	Hold attitude.

TABLE 1.3.1-9 MODAL OPERATIONS (CONTINUED)

PHASE	MODE		ATTITUDE CONTROL REQUIREMENTS	CONTROL SENSORS	CONTROL ACTUATORS	DESCRIPTION OF MODE
Checkout	Earth Point	Develop Orbit Rate about Pitch Axis	Pitch rate = Orbit Rate ($\cong 0.06^\circ/s$) $\pm 10^{-6}$ d/s	R,P,Y rate gyros	R,P,Y 0.1-lb jets	Allowing time to buildup pitch rate to orbit rate, commence at proper moment to buildup pitch rate to orbit rate, so that when pitch rate = orbit rate, spacecraft attitude is such that Instruments point at Earth and Spacecraft X axis points in the direction of light. Update using FHT at each opportunity from this point on.
		Hold orbit rate. Correct for attitude errors.	" " Roll and yaw rates oscillate sinusoidally with magnitude 1°/day, the sinusoid varying at orbit rate. Point at Earth ≤ 0.01 d Rate stability over 30 min $\le \pm 10^{-6}$ d/s Jitter up to 30 sec $\le \pm 1\ \widehat{sec}$ up to 20 min $\le \pm 2\ \widehat{sec}$	R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading.	Hold pitch rate=orbit rate. Continue to update with FHT. Make corrections to rate commands based on updates. Calibrate gyros using FHT updates. Slight oscillatory rate commands are put into roll & yaw axes for orbit regression. An ephemeris calculation is used to compute attitude commands. The DSS is not used for updating when the FHT provides sufficient updates.
Earth-Point Mission	Earth-Point					As described above under checkout.
Stellar Mission	Stellar-Point		Pointing accuracy ≤ 0.01 d. Pointing stability over 30 min $\le \pm 10^{-6}$ d/s. Jitter $\le \pm 2\ \widehat{sec}$. With Instrument signals: Pointing accuracy $\le \pm 0.01\ \widehat{sec}$ Jitter $\le \pm 0.003\ \widehat{sec}$	R,P,Y rate gyros	R,P,Y reactionwheels with magnetic unloading.	A sequence of slews are made to a desired attitude. In the case that the Instrument error signals are used, settling is required. Updates are performed using FHT and Instrument Telescope error signals when available. (Slews between targets are relatively slow for targets scattered in the sky, using the 2 d/m requirement.)

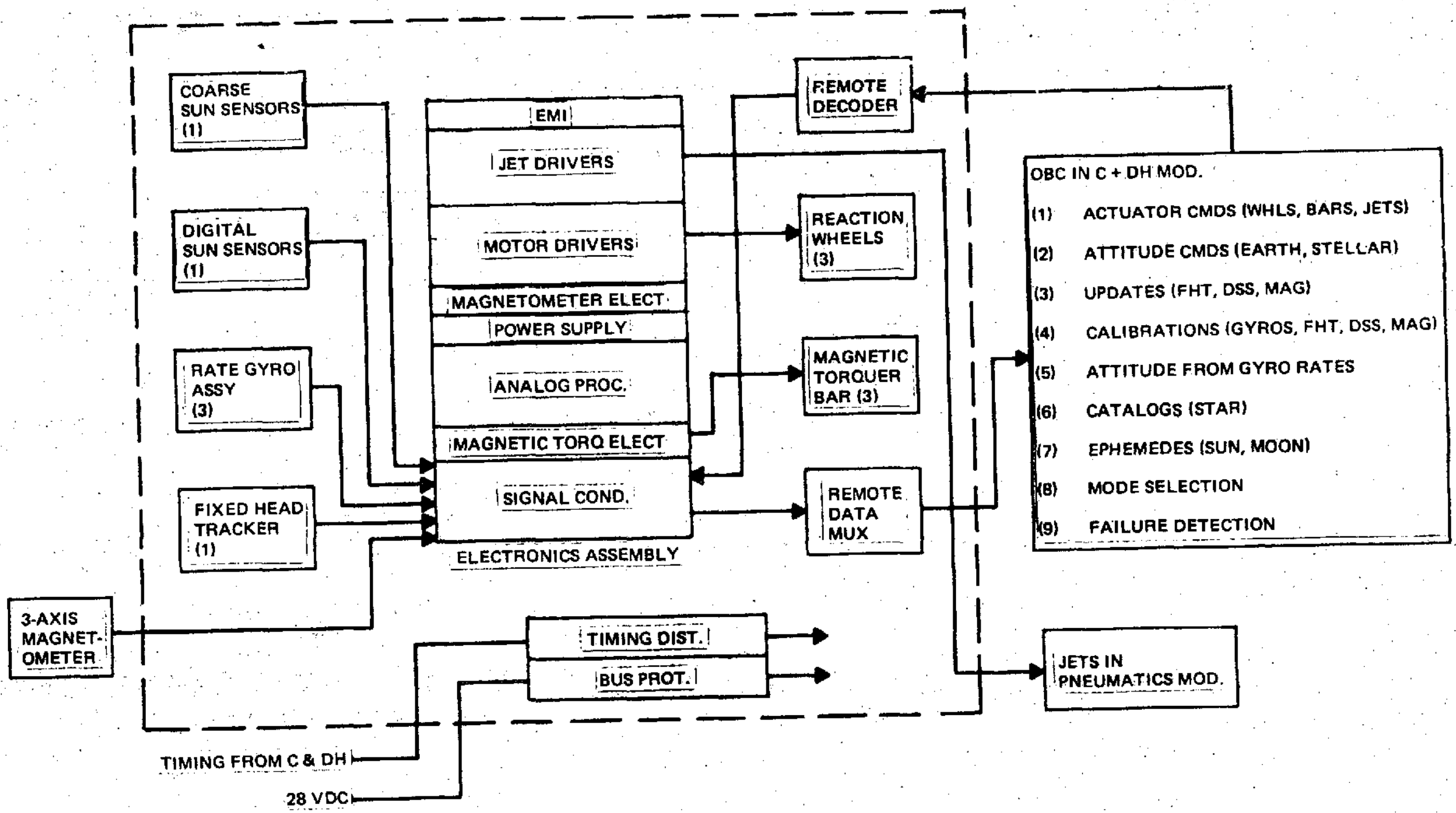


Fig. 1.3.1-1 Block Diagram of Baseline EOS ACS

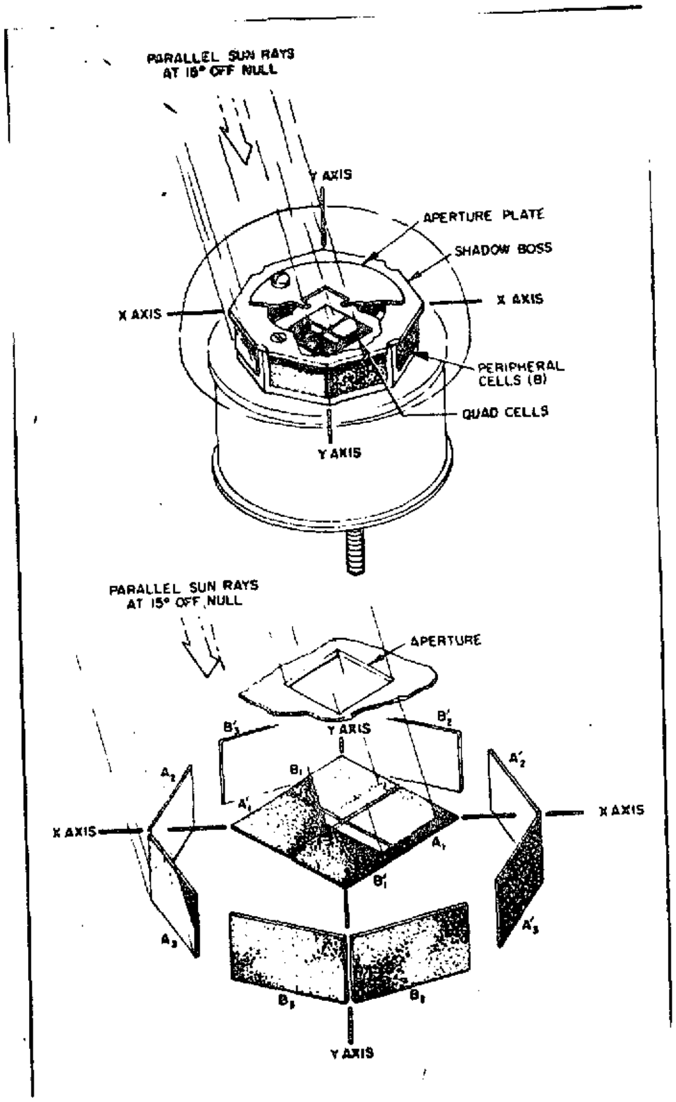
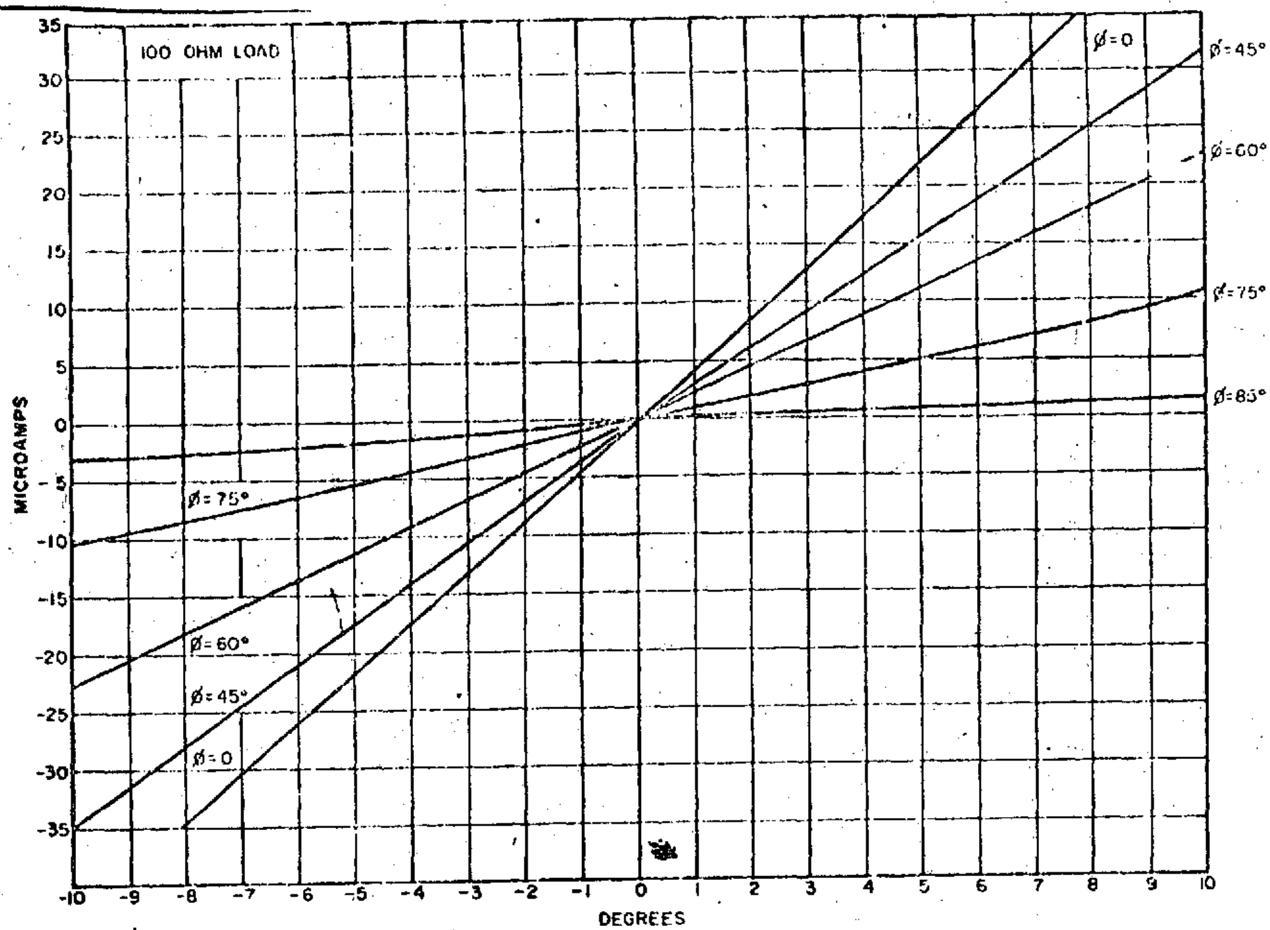
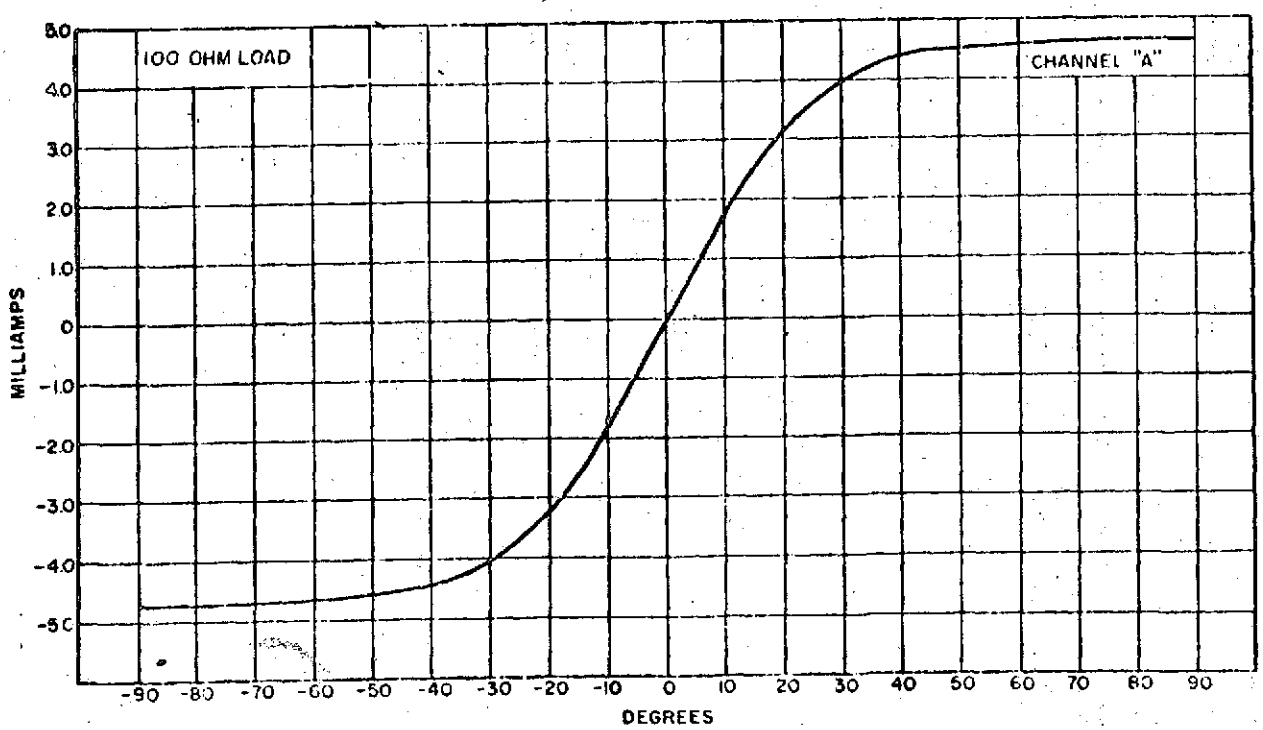


FIGURE 1.3.1-2A BENDIX WIDE ANGLE SUN SENSOR



CURRENT VS ZENITH ANGLE
Illumination—230 Foot-Candles



CURRENT VS ZENITH ANGLE
Illumination—10,000 Foot-Candles

FIGURE 1.3.1-2B BENDIX WIDE ANGLE SUN SENSOR

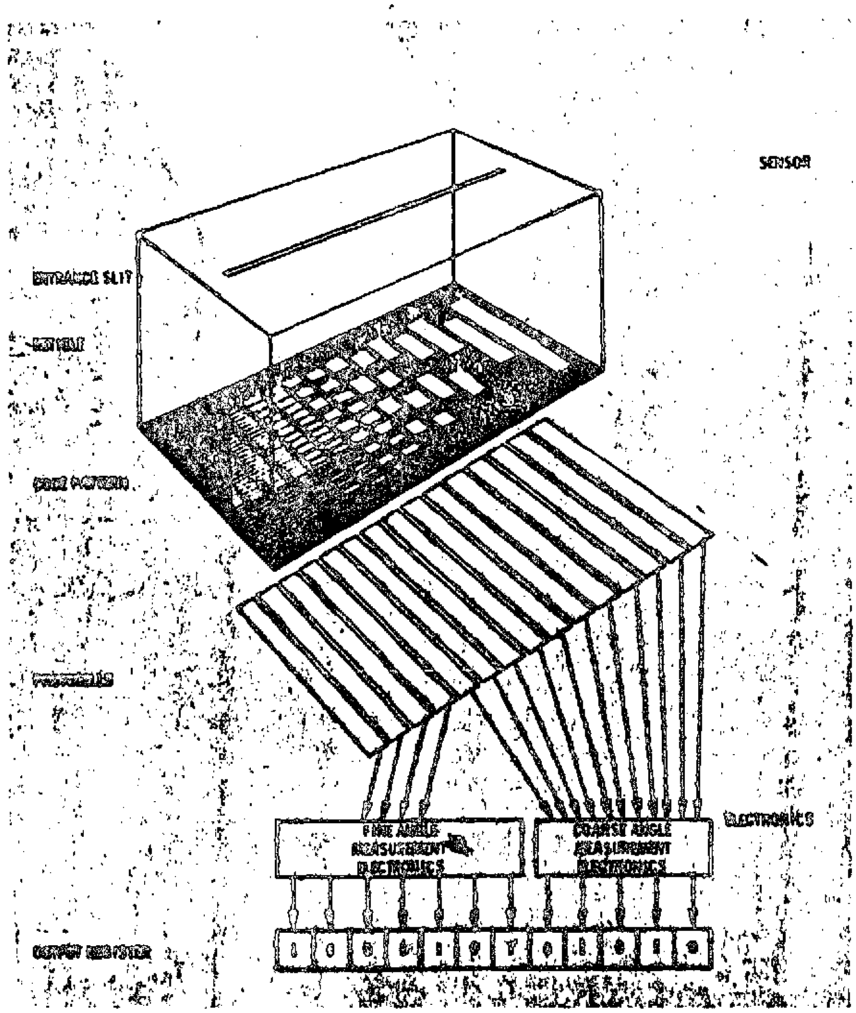


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FIGURE 1.3.1-3 ADCOLE DIGITAL SUNSENSOR

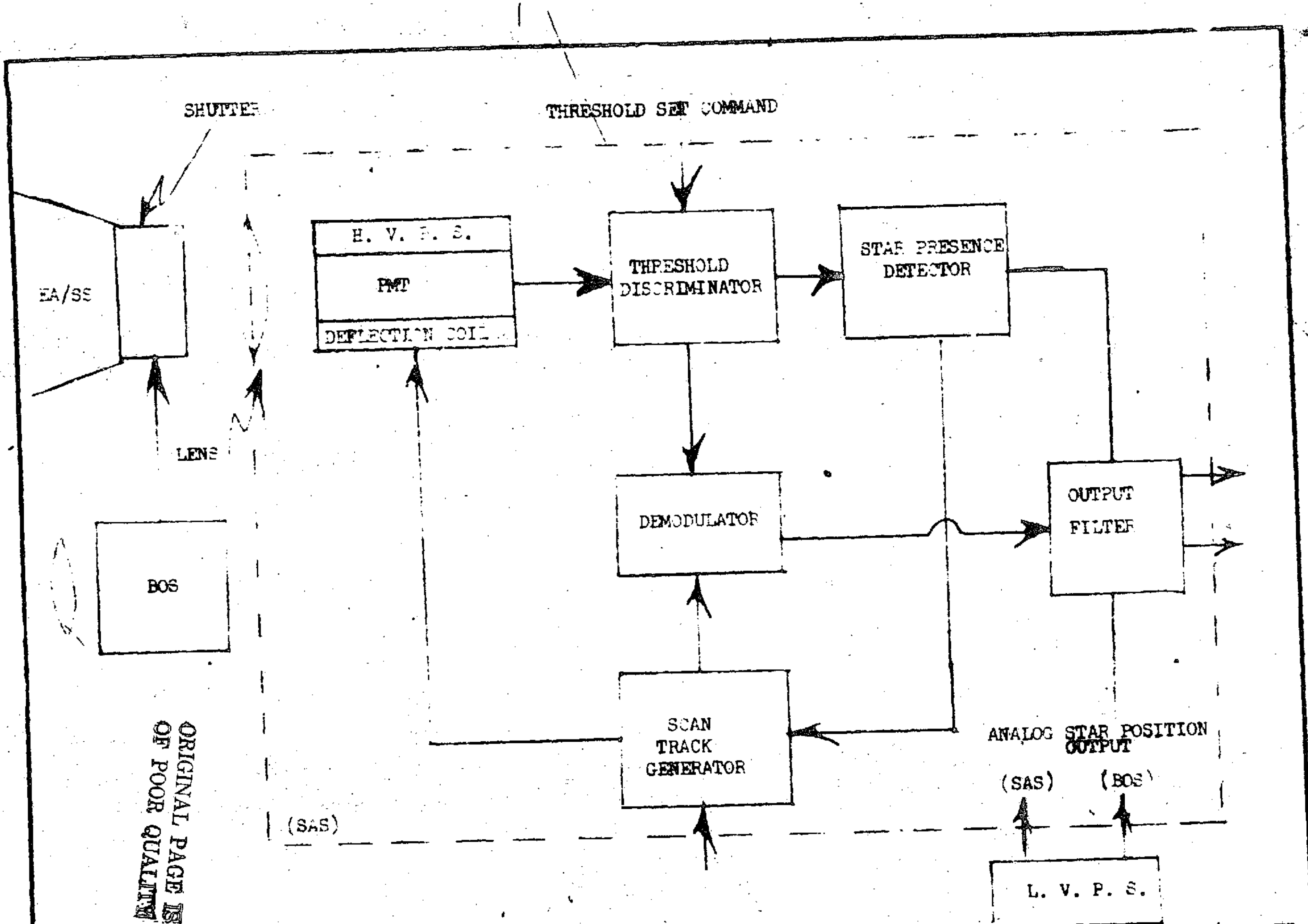


FIGURE 1.3.1-4 BLOCK DIAGRAM OF ITT SRA (FHT)

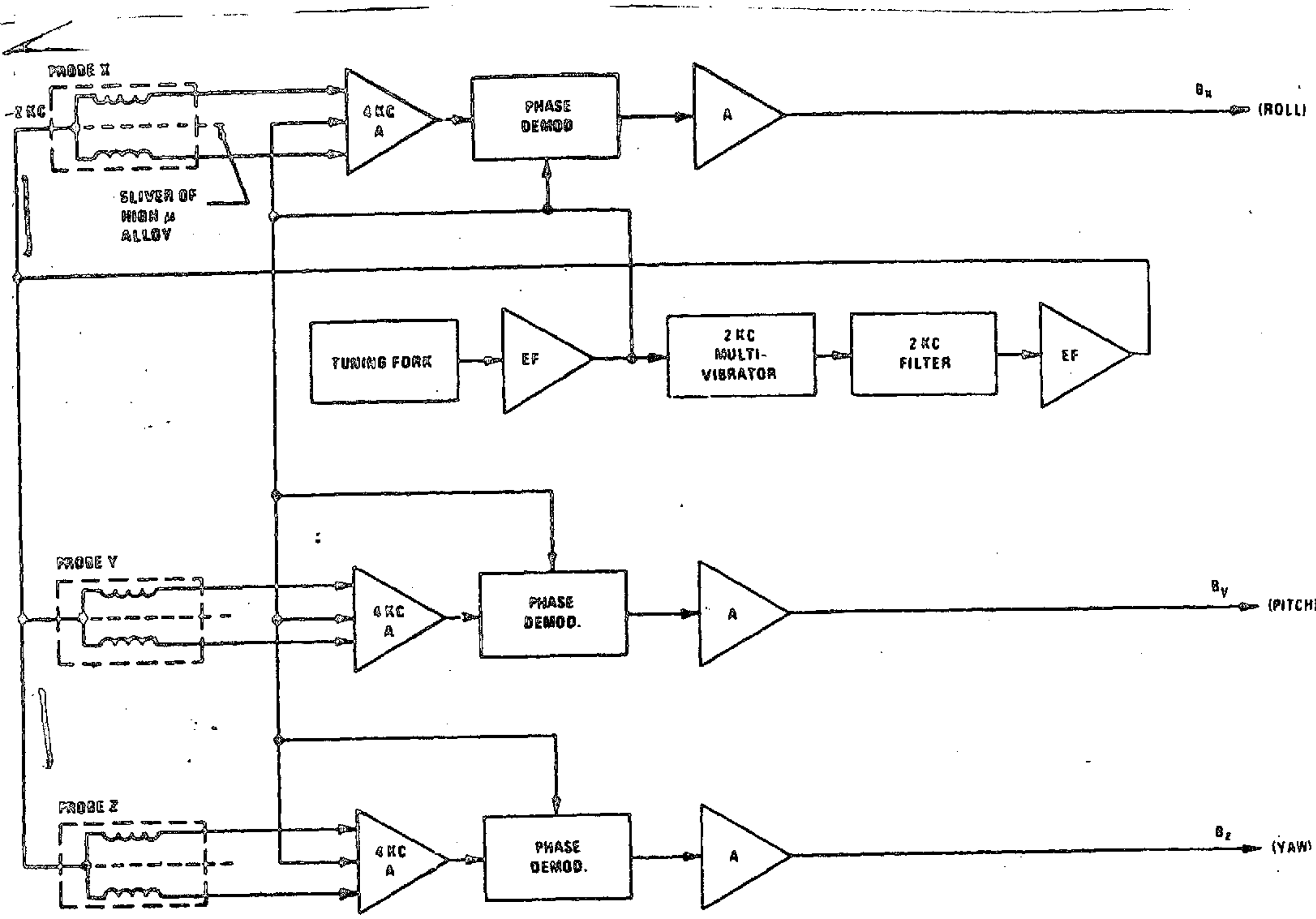


FIGURE 1.3.1-5 **Magnetic Unloading System Magnetometer**

D.1.3.2 COMMUNICATIONS AND DATA HANDLING (C&DH)

D.1.3.2.1 C&DH Subsystem

The C&DH subsystem shall satisfy the EOS requirements (Appendix C, Vol 3) and will be compatible with the operational requirements defined in the GSFC STDN Users Guide No. 101.1 and the GSFC Aerospace Data System Standards X-560-63-2.

The C&DH subsystem shall provide the means of commanding the spacecraft and payload instruments via the uplink, provide onboard data required for ground monitoring of the spacecraft and payload status via downlink telemetry, and transpond ranging signals for ground tracking of the spacecraft. This subsystem shall be located in the Communications and Data Handling Module except for the antennas. The antenna locations will depend on radiation pattern coverage requirements. Other items such as signal conditioning and remote units which are elements of the C&DH subsystem are charged to the module in which they are located and service. The subsystem shall be functionally separate and operate independent of the wide-band communications subsystem.

The general functional requirements for the C&DH subsystem are:

- o Provide telemetry, tracking and command compatibility with STDN, Shuttle orbiter, TDRS (option) and DOI (option)
- o Acquire, process, record, format and route data/commands from/to the appropriate EOS Subsystem Modules
- o Execute ground commands in both real and delayed time
- o Provide on-board sequencing for spacecraft functions scheduled to occur during and after launch prior to initial ground contact
- o Store (on-board) spacecraft housekeeping data between ground contacts (option)

This subsystem shall consist of the C&DH equipment which is composed of a communications group and a data handling group. The communications group is discussed in section D.1.3.2.2 and the data handling group is discussed in section D.1.3.2.3.

D.1.3.2.2 Communications Group

The Communications Group of the C&DH module provides telemetry, tracking and command link compatibility with STDN, Shuttle orbiter, TDRS (option) and DOI (option). Table D.1.3.2-1 tabulates the significant communication link requirements for four link interfaces. Only the interface with STDN at S-Band is fully defined at this time. Table D.1.3.2-1 will be updated to include all of the detail interface requirements for the other links as soon as data is available.

D.1.3.2.2.1 Communications Group Configuration Alternatives

Seven alternate communication configurations were derived. They vary in capability and complexity from the single thread configuration of Fig D.1.3.2-1 to the sophisticated multimode configurations of Figs D.1.3.2-6 and -7. The basic parameters of these alternates are compared in Table D.1.3.2-2 Some of the comparison data is currently not available for configurations 3,4 and 5 since the evaluations of these configurations are still in process.

The primary difference between configurations 1 and 2 is that configuration 1 provides spherical antenna coverage on the uplink and hemispherical antenna coverage on the downlink, whereas configuration 2 provides spherical antenna coverage on both the uplink and downlink. The "A" versions of configurations 1 and 2 have dual redundant transponders.

In configuration 3, an improvement in uplink command reliability is achieved by combining the outputs of receiver/demodulator and selecting the best signal or by cross-strapping the inputs of two demodulators. Results of preliminary analysis indicates that cross-strapping the demodulator inputs is the preferred approach.

Table D.1.3.2-1 EOS Communications Link Requirements

DOWNLINK TELEMETRY PARAMETER	VALUE	COMMUNICATION LINK STDN	TDRS*	SHUTTLE	DOI*
o Frequency	TBD MHz ± .001% (2200 to 2300 MHz)	X	X	X	TBD
o Narrow Band Data Rate	Selectable, 32 Kbps, 16 Kbps, 8 Kbps, 4 Kbps, 2 Kbps, 1 Kbps	X	X	X	X
o Narrow Band Modulation	Split phase PCM/PM on 1.024 MHz subcarrier	X	X	X	X
o Medium Band Data Rate	128 Kbps, 640 Kbps optional	X	N/A	N/A	X
o Medium Band Moudlation	Split phase PCM/PM on carrier	X	N/A	N/A	X
o CCIR Power Flux Density Limits in any 4 KHz Band	-144 dBW/m^2/4KHz	X	X	X	X
	Overhead S-Band Transmission				
o Link Margin	6 dB minimum	X	X	X	X
o Ground Antenna Size	30 ft dish	X	N/A	N/A	TBD
o Gnd System Noise Temperature	125°K	X	N/A	N/A	X
o Bit Error Rate	$\leq 10^{-5}$	X	X	X	X
o Required E/N$_o$	12 dB	X	X	X	X
o Maximum Slant Range	3040 KM	X	TBD	TBD	TBD
o Atmosphere Loss	0.6db	X	N/A	N/A	X
o Polarization	RHCP	X	X	X	X

* Optional Interfaces

Table D.1.3.2-1 EOS Communications Link Requirements (Cont)

UPLINK COMMAND PARAMETER	VALUE	STDN	TDRS*	SHUTTLE	DOI
		COMMUNICATION LINK			
o Frequency	TBD MHz (2025 to 2120 MHz)	X	X	X	TBD
o Command Bite Rate:	2000 bps	X	X	X	X
o Command Modulation:	PCM/PSK - Σ/FM/PM (Uses 70 KHz subcarrier)	X	X	X	X
o Trans Pwr:	10 KW	X	TBD	TBD	X
o Trans. Antenna Size:	30 ft. dish	X	TBD	TBD	TBD
o Link Margin:	6dB minimum	X	X	X	X
o Bit Error Rate:	$\leq 10^{-5}$	X	X	X	X
o Required E/N_o:	12 dB	X	X	X	X
o Maximum Slant Range:	3040 KM	X	TBD	TBD	TBD
o Atmosphere Loss:	0.6 dB	X	N/A	N/A	X
o Polarization:	RHCP	X	X	X	X
Ranging Channel					
o Frequency:	downlink = $\frac{240}{221}$ x uplink	X	TBD	X	TBD
o Ranging Modulation:	PM on carrier	X	X	X	X
o Ranging Technique:	Harmonic tones (500 KHz Max)	X	N/A	X	TBD
	PN	N/A	X	N/A	N/A
o Turnaround Ratio:	221/240	X	TBD	X	TBD
o Group Delay Uncertainty:	$\leq$ 5 nanoseconds	X	TBD	TBD	TBD
o Other parameters:	refer to uplink & downlink requirements				

* Optional Interfaces

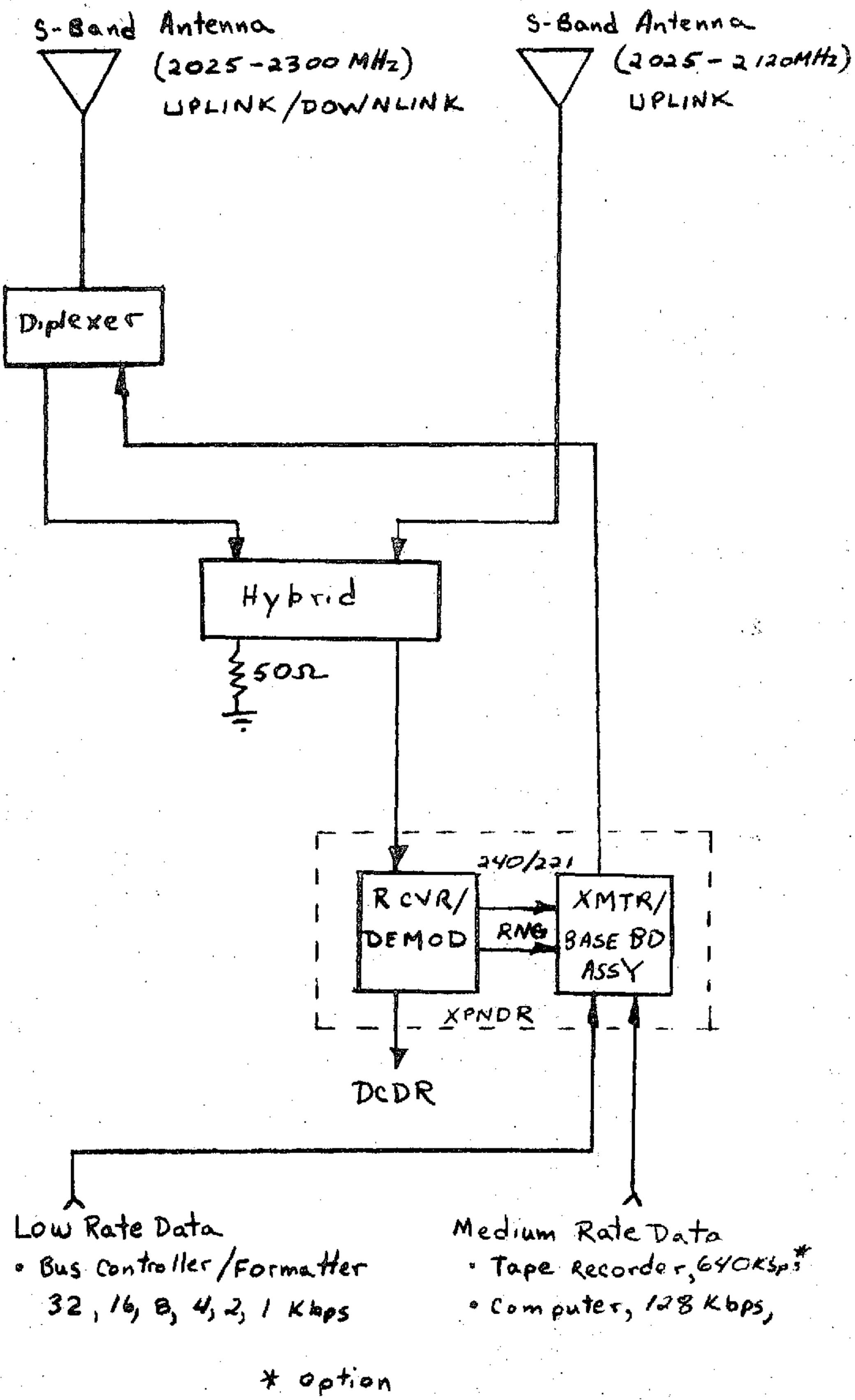


FIGURE D.1.3.2-1 BASIC COMMUNICATIONS CONFIGURATION 1

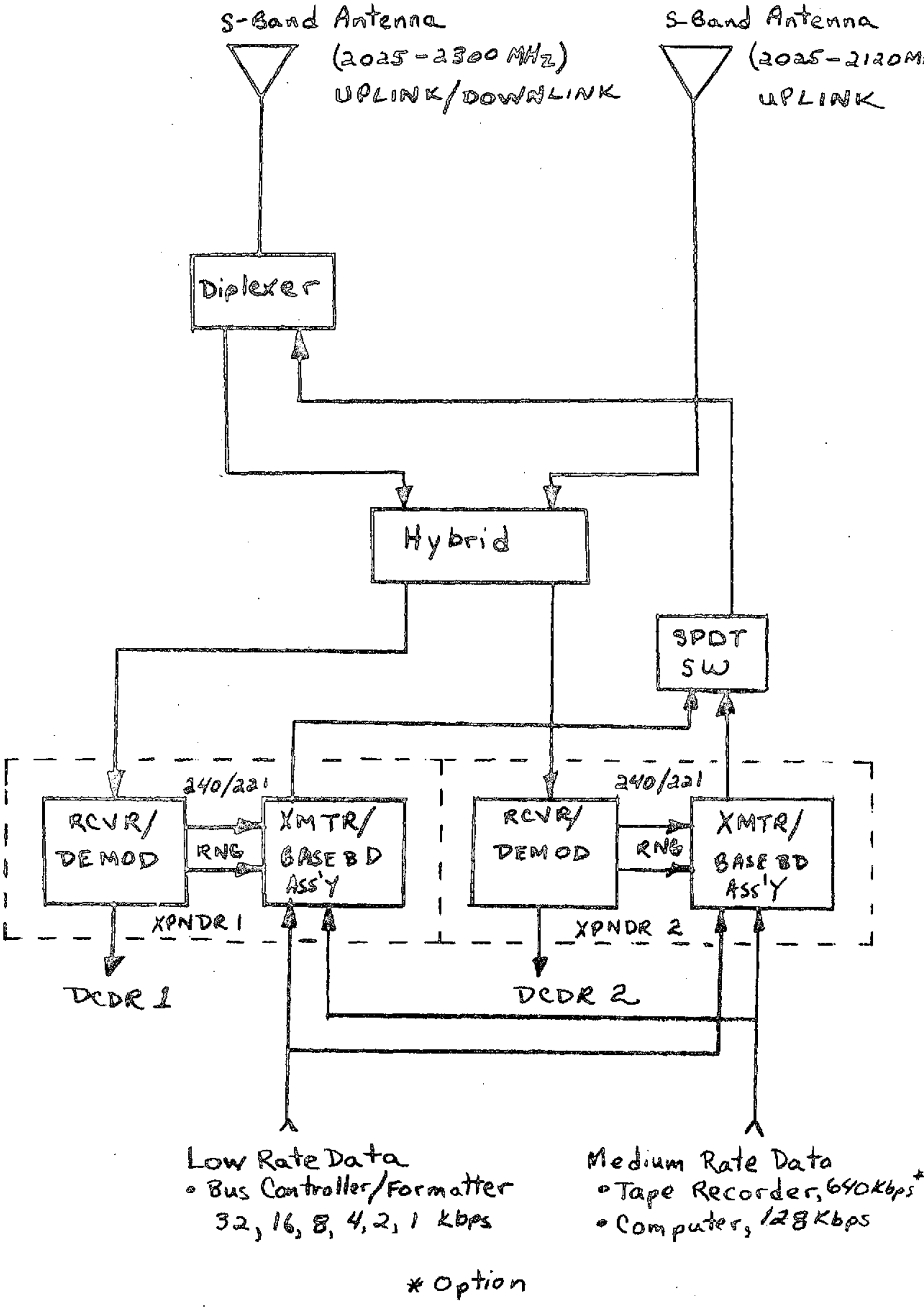


FIGURE D.1.3.2-2 BASIC COMMUNICATIONS CONFIGURATION 1A

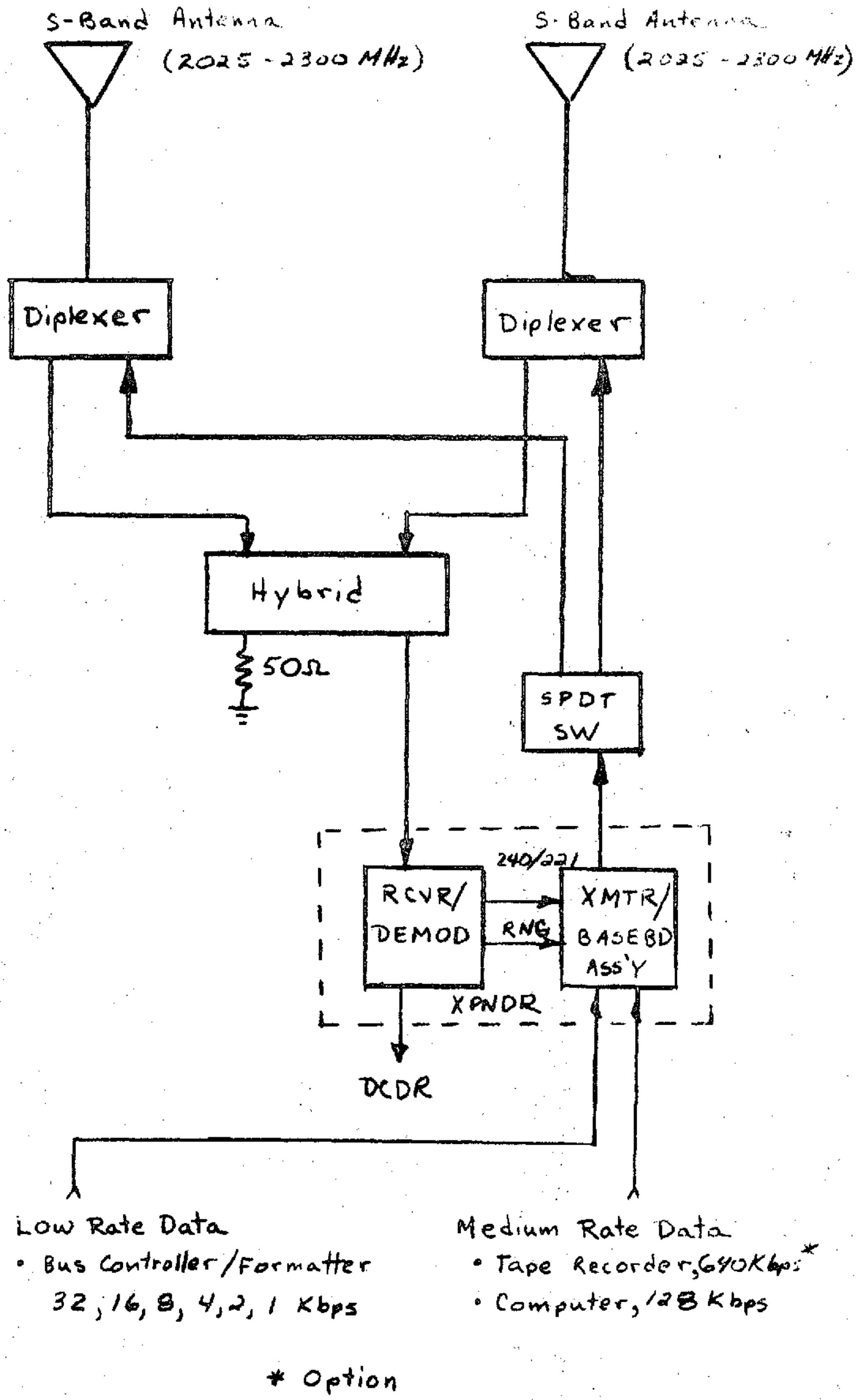


FIGURE D.1.3.2-3 BASIC COMMUNICATIONS CONFIGURATION 2

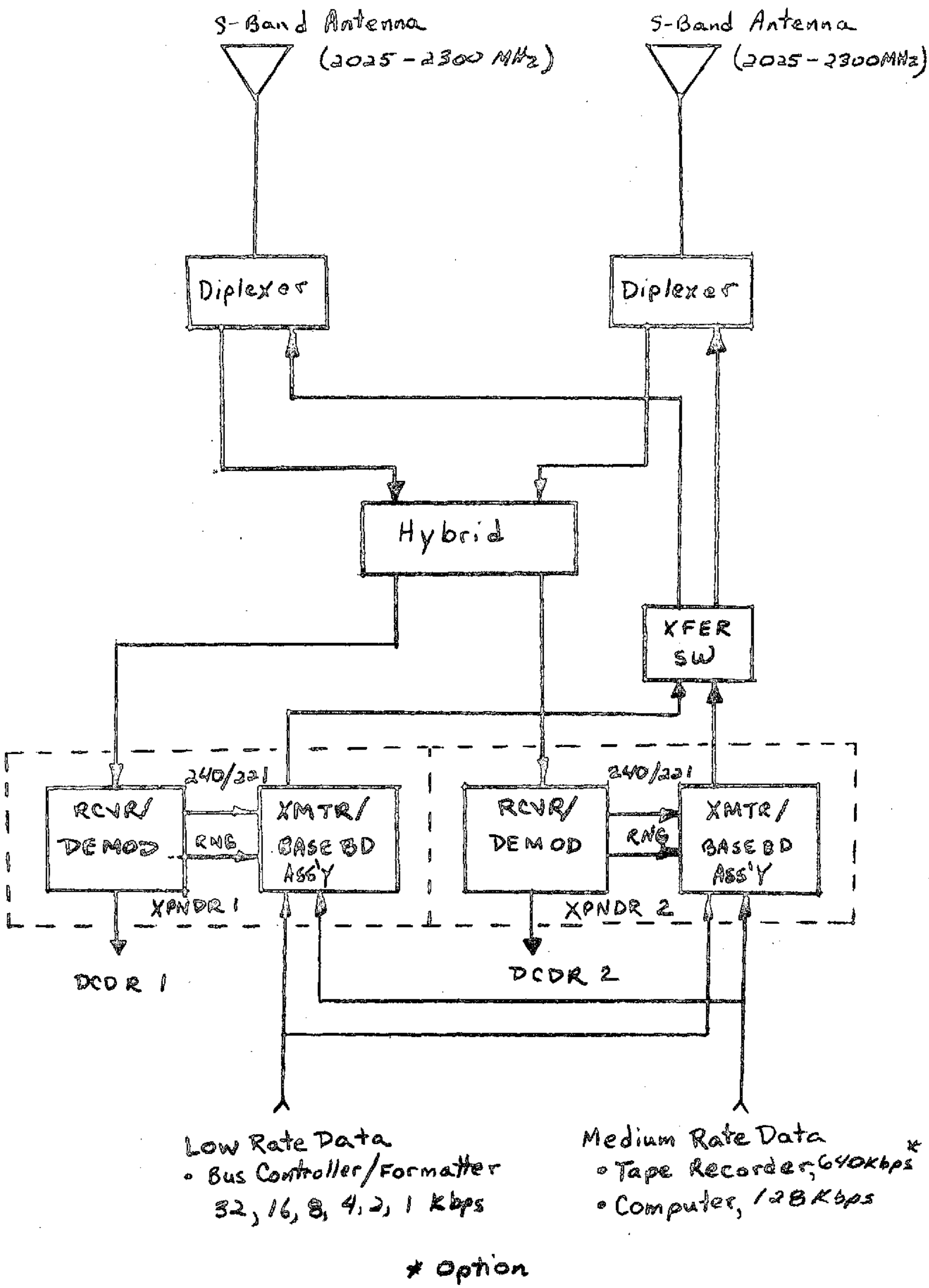


FIGURE D. 1.3.2-4 BASIC COMMUNICATIONS CONFIGURATION 2A

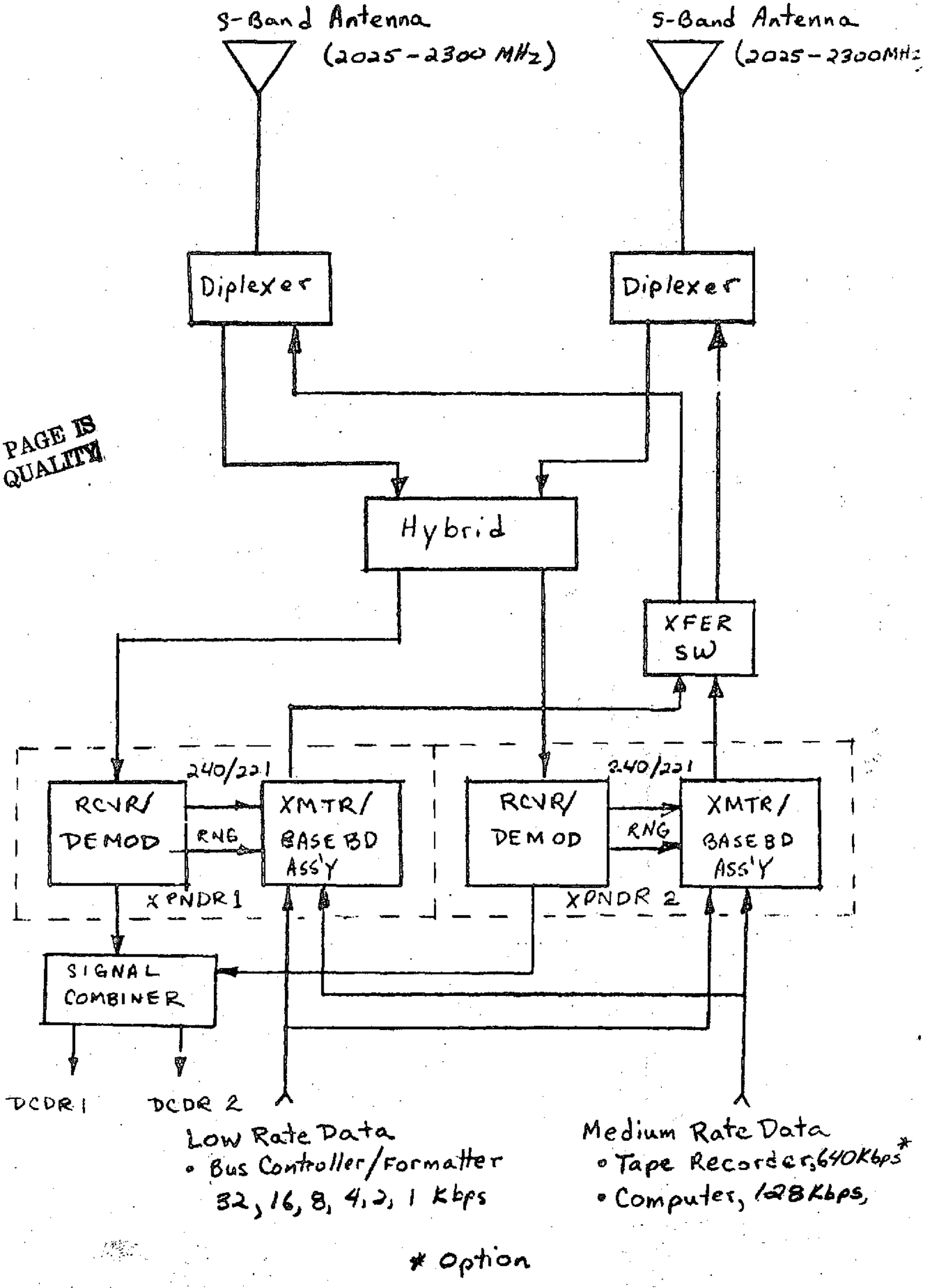


FIGURE D.1.3.2.-5 BASIC COMMUNICATIONS CONFIGURATION 3

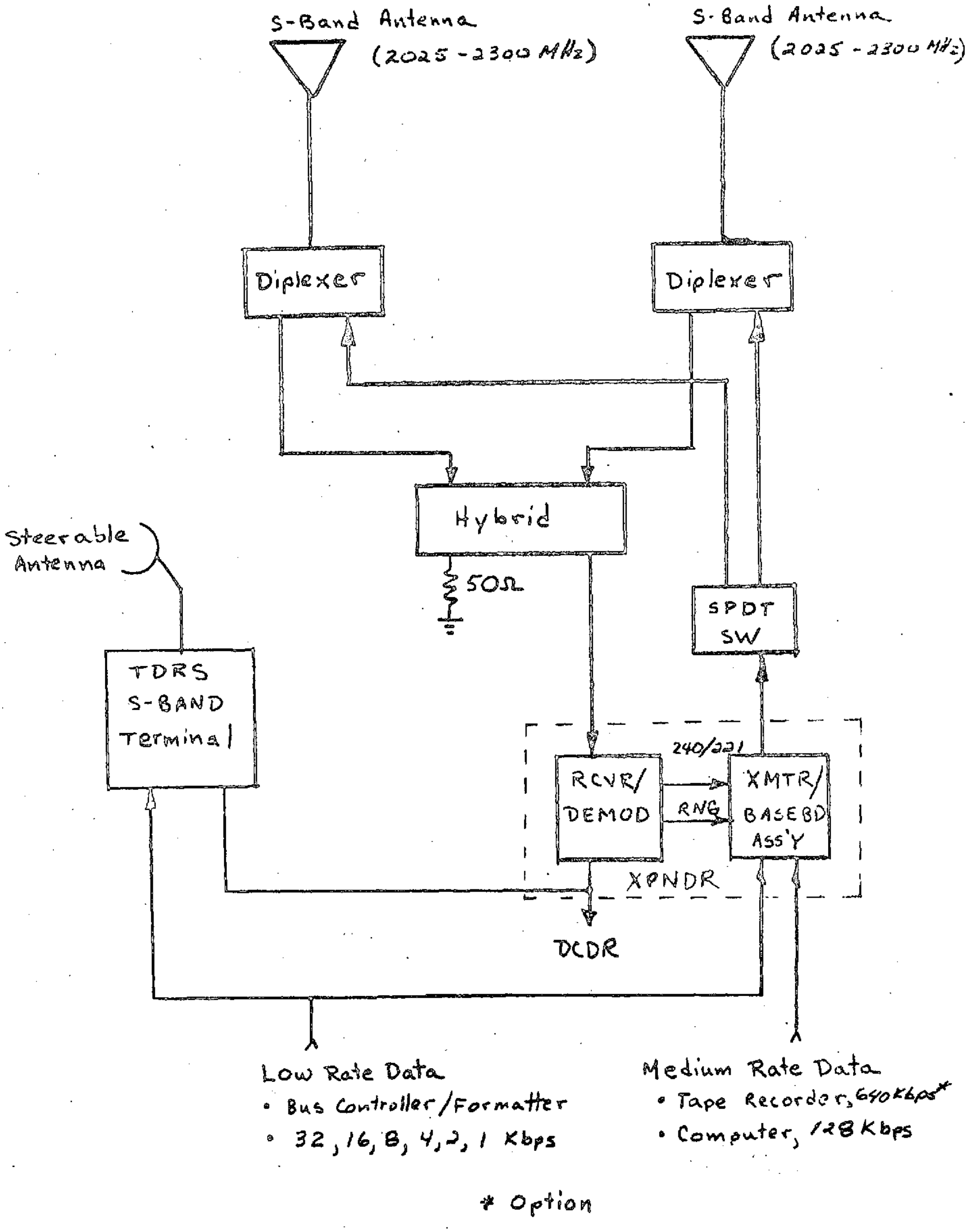


FIGURE D.1.3.2.-6 BASIC COMMUNICATIONS CONFIGURATION 4

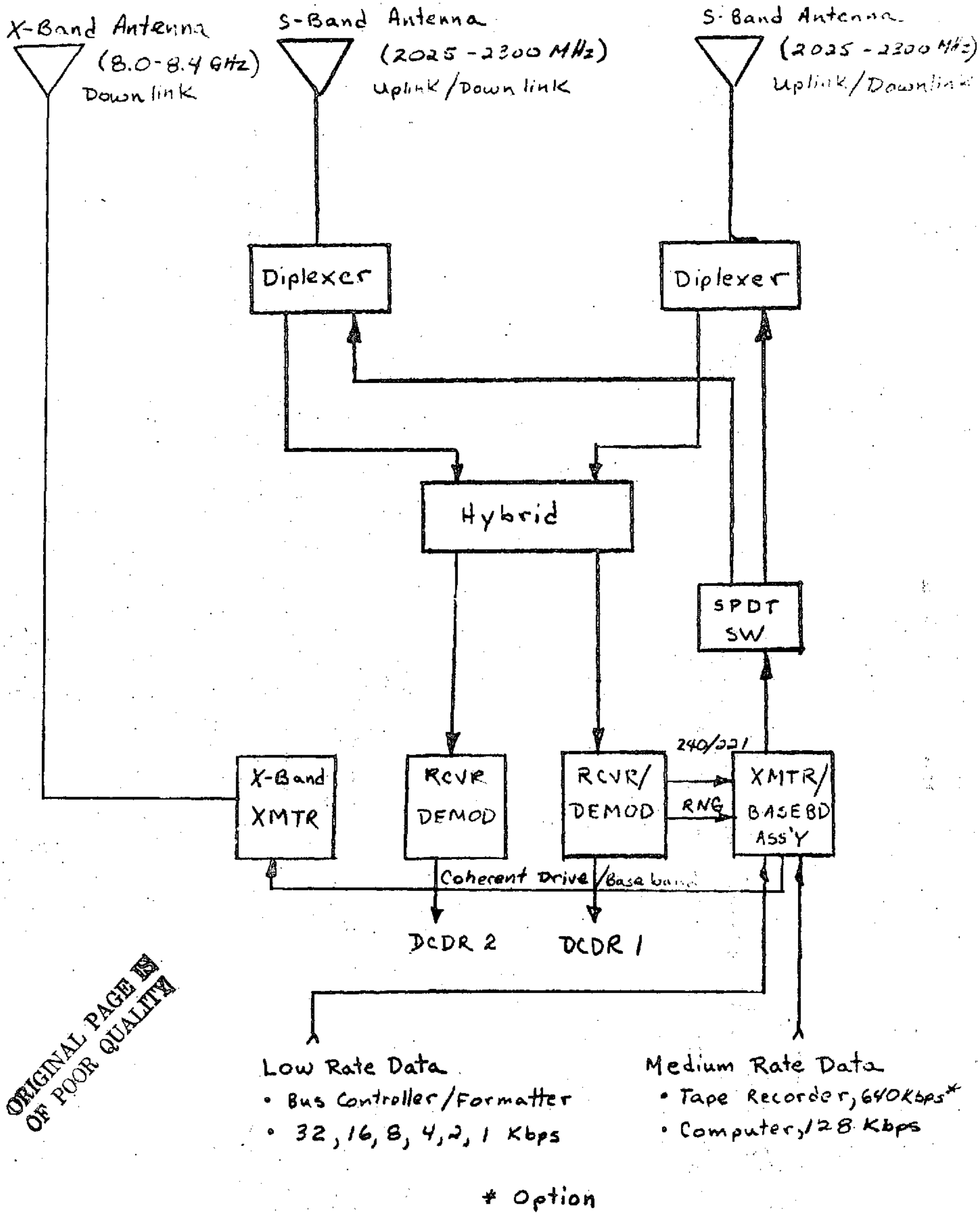


FIGURE D.1.3.2.-7 BASIC COMMUNICATIONS S/X BAND CONFIGURATION 5

TABLE D.1.3.2-2

COMMUNICATION CONFIGURATION COMPARISON

	CONFIGURATION						
PARAMETER	1	1A	2*	2A	3	4	5
o S-Band Antenna Coverage							
Spherical Uplink/Downlink-1 Ant	X	X					
Spherical Uplink/Downlink-2 Ant			X	X	X	X	X
o Dual Redundant Transponders		X		X	X		
o Signal Combining or Cross-strapping					X		
o TDRS Interface						X	
o X-Band Downlink							X
o Weight (lbs)	9.5	14.2	10.2	14.9	TBD	10.2+ TDRS	TBD
o Power (Watts)	12.0	14.5	12.0	14.5	TBD	TBD	TBD
o Cost ($K)	283	390	274	381	TBD	TBD	TBD
o Risk	Min	Min	Min	Min	Mod	Max	Min
o Spacecraft Integration Complexity	Min	Min	Min	Min	Min	Most	MOD

* Selected Configuration

Configuration 4 is configuration 2 plus a TDRS S-Band terminal. The terminal includes a S-Band transceiver package and a steerable antenna. Since the wideband communications (refer to appendices to this book) will also have an interface to the TDRS at Ku-band, a dual frequency S/Ku-Band steerable antenna is being considered to satisfy both the narrow band and wideband communications requirements. Portions of the TDRS S-Band transceives (i.e., receiver front end, transmitter) may be co-located with the steerable antenna to reduce RF losses while other portions (i.e., demodulator, baseband assembly) may be located in the communications and data handling module. The steerable S-Band antenna (7 to 11 ft. diameter requirement) will probably be located on a boom to minimize blockage problems.

The last configuration 5, provides a downlink in the 8.0 to 8.4 GHz frequency band allocated for operational earth resource satellite programs. The uplink command would still be at S-Band. A downlink capability would be retained at S-Band in order to provide NASA with maximum command and control capability of the EOS spacecraft from all STDN ground stations.

The prime ground station for this X-Band downlink would be the Department of Interior (DOI) station at Sioux Falls. The S-Band transponder has a phase coherent output available for driving an X-Band transmitter, enabling a smooth transition to this configuration. Dual redundant S-Band receivers are incorporated in this configuration to increase uplink reliability.

It is possible that configurations 4 and 5 may be combined to provide interfaces with STDN, TDRSS and DOI. In any case, the performance/design requirements described in Table D.1. 3.2-1 consider this possibility and the derived alternate configurations will enable any combination of options to be costed.

The alternative configurations have the following types of components:

- o Transponders
- o Transmitters
- o Receivers
- o RF Coaxial Switches
- o Diplexers
- o RF Couplers (Hybrids)
- o Antennas

Candidate components are compared versus significant technical features and procurement costs in Table D.1.3.2-3.

Two primary approaches for the antennas were considered, one using a broadband antenna capable of operating over the full 2025 to 2300 MHz band versus a narrow band antenna tuned to a discrete frequency within that range. The broadband approach was selected for the baseline configuration since all antennas required for EOS could be procured at a single time to minimize cost and since the narrow band microstrip antenna approach would require four S-Band antenna per vehicle, (i.e., separate transmit and receive antennas).

An integrated S-Band Transponder/Antenna multiplexer design, e.g. ERTS or JBS (Japanese Broadcast Satellite) S-Band Transponder assembly, can be compared to the separate electronics packaging design similar to the ELMS Tracking, Telemetry and Command Subsystem using Table D.13.2-3. The integrated electronics approach using the Motorola transponder assembly designed for the JBS results in minimum weight and input power. The weight differential is further increased when the weight of cables and connectors required for the separate electronics approach is considered. Volume is essentially the same for either approach.

Table D.1.3.2-3

Candidate Communication Group Components

Integrated Electronics	Mfr.	Previous Program	Wt. (lbs)	Av Pwr (watts)	Vol (in^3)	Cost ($K) N.R.	R	Total	Comments
S-Band Transponder	Motorola	JBS	12.5	5/14.5*	563	141	214	355	Dual Unit
	Motorola	JBS(MOD)	7.8	2.5/12*	253	141	107	248	Single Unit
	Motorola	ERTS	25.0	7/23*	624	115	175	290	Dual Unit
	Cubic								
	Cincin. Elect.		Request for quotes in process						
	Philco Ford								
Separate Electronics									
S-Band Transmitter/ Baseband	Teledyne	ELMS	3.5	33.6	72	19	20	39	
	Conic	LCRU	2.5	34.0	51	35	33	68	ELMS quote
S-Band Rcvr/Demod	Cincin. Elect.	ELMS	3.6	5.0	93	40	34	74	
S-Band Hybrid	Wavecom	ELMS	1.0	-	8	6	2	8	
	Sander Assoc.	F-14	0.5	-	8	72	2	80	GPS quote
S-Band Diplexer	Wavecom	LCRU	1.0	-	27	10	3	13	
S-Band Switch	Transco	ERTS	0.1	-	0.8	6	2	8	
Antennas									
Broadband	GE	ELMS	1.2	-	50	33	8	41	ELMS quotes
	RCA	Viking	0.8	-	12	113	36	149	
	GAC	F-14	1.5	-	40	49	9	58	
Narrowband	Ball Bros	ELMS	0.5	-	1.3	15	2	17	

* Rx/Rx + Tx

These previous factors indicate that the integrated S-Band Transponder is the preferred approach, however, a cost comparison is required to determine the final selection. The Motorola cost data for the integrated transponder was derived from a responsive quote and cost data for separate electronics are based on the ELMS program. Additional quotes for the integrated transponder and separate electronics designed to EOS requirements are still in process. A review of Table D.1.3.2-3 shows that separate electronics appears to have lower recurring and non-recurring costs ($81K vs $141K and $62K vs $107K respectively). However, additional program cost factors must be considered which increase the cost of the separate electronics approach.

These costs are associated with the procurement of five items versus that for one procurement, the additional costs for integrating (physical and electrical) the separate electronics, and the additional costs for modifying the ELMS transmitter/receiver for coherent operation and compatibility with STDN. These cost factors are considered sufficient to make the separate electronics approach more expensive than the integrated and therefore the latter is selected for the communications configuration. Additional quotes in process are expected to confirm this conclusion.

D.1.3.2.2.2 <u>Selected Communications Group Configuration</u>

Configuration 2, Fig. D.1.3.2-3 is the selected basic communications configuration. A single S-Band transponder unit with integrated diplexers, hybrid and coaxial switch is utilized in conjunction with two broadband S-Band shaped beam antennas. It satisfies the functional and performance/design requirements shown in Table D.1.3.2-1 for the STDN S-Band interface.

This baseline is considered a low risk design because it uses space proven off-the-shelf components. Minimal non-recurring costs are attributed to documentation, program management and minor design changes to satisfy EOS program requirements. The equipment list and cost allocation for the selected configuration is shown in Table D.1.3.2-4.

Primary communication modes of operation provided for the downlink and uplink are:

<u>Downlink</u>

- o Ranging
- o Ranging and Narrow Band Data
- o Medium Band and Narrow Band Data
- o Narrow Band Data Only
- o Medium Band Data Only

<u>Uplink</u>

- o Ranging
- o Ranging and Command
- o Command only

TABLE D.1.3.2-4 SELECTED COMMUNICATIONS GROUP CONFIGURATION

COMPONENT	QTY/VEH	WT. EA. (LBS)	VOL. EA. (IN^3)	PWR EA. (WATTS)	STATUS	SOURCE	COSTS (PER VEH) NR	R
S-Band Transponder Assembly								
o Transponder	1	7.8	253	12W(R_x&T_x)	M	Motorola/ JBS	$141K	$107K
o Diplexers	2			2.5 W (R_x)				
o Hybrid	1							
o Coaxial Switch	1							
TLM/CMD Antennas	2	1.2	50	N/A	E	GE/ELMS	$10K*	$16K
Coaxial Cable Assemblies	2	TBD	TBD	N/A	E	TWC/ELMS	-	$0.2K

* Lower NR costs than ELMS because Qual. testing is not required.

The narrow band data is transmitted on a 1.024 MHz subcarrier while the medium band data is modulated directly on the carrier. Ranging is accomplished by coherently transponding harmonic tones consistent with GSFC ranging equipment specification S-813-P-19. Uplink commands are accomplished on a 70 KHz subcarrier which is compatible with the NASA/STDN Spacecraft Command Encoder (SCE).

D.1.3.2.2.2.1 TLM/CMD Antennas

Two identical antennas are utilized to provide spherical pattern coverage for the S-Band telemetry and command links. The antennas are mounted on opposite sides of the spacecraft as shown in Fig. 2.9.1.1.10-1. The antenna selected was originally developed by General Electric for the NASA ERTS satellite. The version to be used on EOS is identical to the antenna procured by Grumman from GE for the USAF ELMS satellite. The ELMS version of this antenna has two significant design improvements, thermal control paint which will permit its mounting on surfaces that see direct sun light and steel inserts in radiating elements to permit vibration at higher vibration levels. The delta-qualification program on ELMS consisting of humidity, vibration and RF power handle at pressures between sea level and 10^{-5} torr have been successfully completed.

The antenna is a modified turnstile, with its radiating elements tilted downward approximately 30 degrees with the element height above an 8 inch diameter reflector/base adjusted to provide a shaped beam radiation pattern. A minimum gain with respect to a right hand circularity polarized isotropic radiator of 0db on axis increasing to + 3db at 40 to 65 degrees off axis and decreasing to 0 db at 78 degrees off axis is provided.

D.1.3.2.2.2.2 S-Band Transponder Assembly

The S-Band transponder assembly is a derivative of the Motorola M-Series transponder line. It performs the following functions:

- Phase locks to and tracks the frequency of an uplink S-Band RF signal
- Demodulates and extracts command and ranging information from the uplink signal
- Generates in the presence of an uplink signal, a S-Band transmit carrier which is related to the uplink signal by a frequency ratio of 240/221
- Generates, in the absence of an uplink signal a stable S-Band transmit carrier which has been derived from an internal crystal oscillator
- Phase modulates the downlink carrier with a composite baseband signal consisting of telemetry and ranging information
- Provides status and telemetry information to the spacecraft telemetry system
- Includes an integrated microwave antenna multiplexer unit consisting of two diplexers, a hybrid power splitter and a SPDT coaxial switch

The selection of this transponder assembly was based on several factors. They are:

- Takes advantage of recent technology which is being developed for the JBS program
- Maximum use of space-proven designs
- Excellent overall temperature stability
- AGC Loop on ranging channel eliminates video limiting and resultant group delay variations
- Minimal ranging delay variation of $\pm$ 10 nanoseconds total for temperature and input signal variations

- o Lower weight compared to ERTS Transponder Assembly or individual components
- o A phase coherent output available for driving an X-band exciter applicable for a narrow band downlink to a DOI terminal
- o Space qualified mechanical packaging design capable of housing two transponders
- o Cross-strapping capability in dual redundant configuration allowing either receiver to drive either exciter

Low rate data and medium rate data are combined in the baseband assembly located in the transponder unit. The turnaround/tone ranging signal is also combined in the baseband assembly for downlink transmission either with low rate data or alone. The medium band composite baseband signal phase modulates the transmitter and is amplified to a level of 2 watts at S-band. In the low data rate mode only, the transmitter output power is decreased to 0.2 watts to meet the CCIR spectral flux density requirements.

The transmitter can be connected to either antenna through a SPDT coaxial switch in order to provide spherical antenna coverage on the downlink.

For the uplink, the receiver/demodulator is always powered on and capable of detecting uplink command signals. RF switching is not used in the uplink to increase reliability. The two receive output channels of the diplexers are connected to the receiver via a RF power splitter (Hybrid). A 70 KHz demodulator is incorporated into one module of the transponder. The demodulator output feeds a command decoder.

D.1.3.2.3 DATA HANDLING GROUP

The Data Handling Group (DHG) must acquire, process, record, format and route data/commands from/to the appropriate EOS Subsystem/Module (Communications, ACS, Elec. Pwr, Orbit Adjust and Transfer, etc.) and the support vehicle (e.g., Shuttle Orbiter, etc.). In addition the group shall perform the required attitude control computations issuing the necessary commands, receive commands from the ground and distribute/execute these in real time or store them for delayed execution on a time or event basis.

D.1.3.2.3-1 DATA HANDLING GROUP REQUIREMENTS

Detailed DHG requirements and their origin are outlined in Table D.1.3.2-5. The DHG shall be capable of transmitting to Shuttle Orbiter or ground (via comm.) variable data rates of 32/16/8/4/2/1 KBPS and receiving 2 KBPS in commands. Commands uplinked from ground are 40 bits in length, 24 bits of which are defined by NASA as the computer data word thereby requiring two locations in storage in any 16 to 18 bit word length computer.

GAC software sizing estimates (see Table E.3.2.9-2) for command storage, spacecraft control systems monitoring, etc. define 23.3K (eighteen bit) words including margin are required for storage in the computers main memory. Resolution to 30 meters is required for MSS image processing. Twenty four bit word length provides resolution to seven meters while still accommodating earth orbit dimensions with margin. Throughput requirements range from 6-13KOPS (Kilo Operations per Second). The RGA Service routine is the main driver utilizing 3 KOPS.

The GAC measurements list (see Enclosure D.1.3.2-1) identifies and codes each measurement and defines its signal type (temperature, pressure, discrete etc.) sample rate, high-low value, accuracy and whether it is hardwired (H) for GSE (Ground Support Equipment), telemetered (T) or used on-board (O). The list is partitioned per module and then per the equipment within each module. Measurements which vary as a function

Table D.1.3.2-5 Data Handling Subsystem Requirements

ITEM	REQUIREMENT	ORIGIN
COMPUTER		
MAIN MEMORY (K WORDS)	23.3	GAC SOFTWARE SIZING ESTIMATE
WORD SIZE (BITS)	18-24	NASA, GAC SIZING ESTIMATE
THRUPUT (KOPS)	6-13	GAC SFW SIZING ESTIMATE
LANGUAGE	ASSEMBLY	GAC MEMORY EFFICIENCY
CLOCK STABILITY	±1 PART IN 10^6 per day	NASA
TLM RATES (KBPS)	32/16/8/4/2/1	NASA GSFC & JSC ORBITER
CMD RATES (KBPS)	2, 2.4	NASA GSFC & JSC ORBITER
REMOTE UNITS (#/SPACECRAFT)	7	GAC MEASUREMENTS & COMMANDS SIZING
MEASUREMENTS	240	GAC MEASUREMENTS & COMMANDS SIZING
COMMANDS	160	GAC MEASUREMENTS & COMMANDS SIZING
CAUTION & WARNING FUNCTIONS	9-12	GAC SIZING (LAUNCH VEHICLE DEPENDENT)
TAPE RECORDER (OPTIONAL)		
CAPACITY (MBITS)	10^6	GAC
RECORD RATES (KBPS)	32/16/8/4/2/1	NASA
REPRODUCE RATE (KBPS)	640	NASA
RECORD TIME (MINUTES)	560	GAC

of launch vehicle configuration (OA&T Module)
are also identified. This identification and Module/equipment partitioning makes possible rapid assembly of a measurements list for a particular EOS mission and launch vehicle configuration.

The measurements list permits accurate determination of the data acquistion requirements such as the number of remote units required per module and spacecraft (seven), the amount and type of signal conditioning, A/D (Analog to Digital) conversion and sensors required for each EOS. Sample rates of Telemetered Measurements are summed to define the required EOS telemetry rates. Table D.1.3.2-6 summarizes the present estimated number of measurements types and commands required per EOS module. Figure D.1.3.2-8 defines the number of remote units data and command channels required per EOS module. This figure shows that if the maximum number of remote unit channels are to be fixed 64 channels represents on efficient selection.

The basic spacecrafts approximately 240 measurements and 160 commands are handled by five remote units (64 inputs and 64 outputs each) while two more remotes are dedicated to the Instruments. The TM, HRPI, and MSS all require approximately 118 measurements, plus 48 discrete commands and 4 instruction words each. The MUMS is estimated to require 16 measurements and 16 discrete commands.

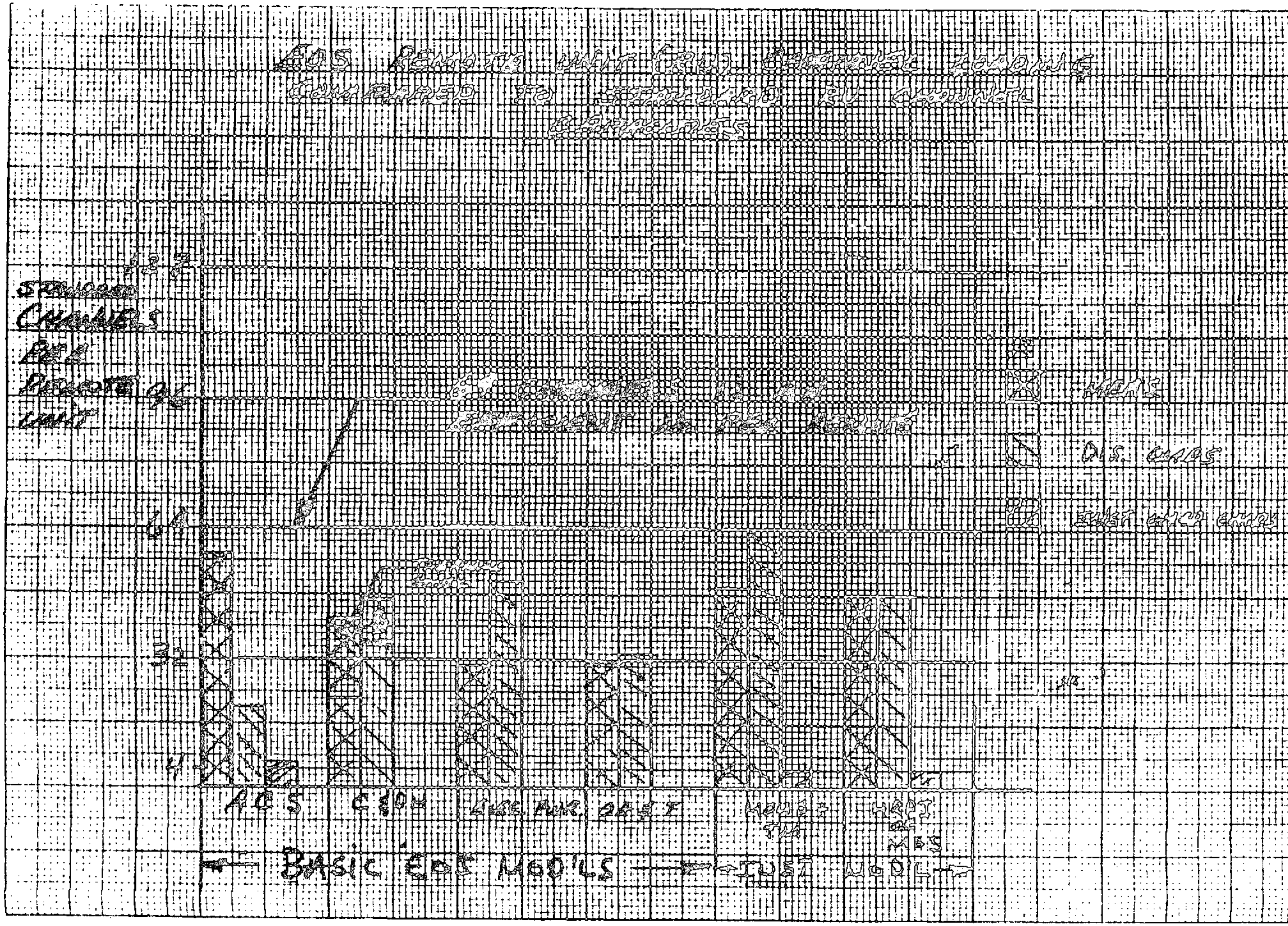
Recording requirements are driven by telemetry line data rates; the maximum time EOS is out of ground contact 5-7 hours (GAC estimates based on GAC mission trajectory analysis results of EOS Sun Synchronous mission) and time (11 minutes max.) EOS is in ground contact following such a period.

Table D.1.3.2-6 EOS Measurements and Commands Summary

MODULE \ TYPE		MEASUREMENTS: PRESS.	TEMP	DISCRETE	VOLT.	CURRENT	RATE	ANG.	WORD	MAG/GAUS	TOTALS	MUX CHS.*	CMDS DIS/SER
ACS		3	11	47	7		6	9	5	1/3	92	56	20/6
C & DH			4	13	10				12		59	44	36/0
INCLUDING THERMAL EXTERNALS			20										10/0
ELEC POWER			10	24	7	8		2			51	30	52/0
OA & T		2	24	9	3						38	31	34/0
BASIC S/C TOTAL		5	69	93	27	8	6	11	17	4	240	N/A	152/6
INSTR-UMENTS	MUMS (AND)			16							16	50	16/0
	TM		9	80	6	3			20		118		48/4
	HRPI (OR)		9	80	6	3			20		118	48	48/4
	MSS		9	80	6	3			20		118		48/4
INST. TOTAL			27	256	18	9			60		370	N/A	160/12

* GROUNDRULE PACK 8 DISCRETES TO A MUX CHANNEL

FIGURE D.1.3.2-8



While EOS is attached to Orbiter, the Orbiter crew must be alerted and have the capability of monitoring any EOS parameters which will indicate a potentially hazardous condition. Alerts will be monitored by the flight crew at the flight crew stations and the specialist at the Orbiter Mission Specialist Station (MSS).

The EOS Data Handling Group (DHG) will be "on" during the Orbiter Launch, Boost, Ascent and Descent (retrieve) or resupply phases. Caution and warning functions will be multiplexed to the Orbiter by the DHG and also hardlined to the Orbiter while EOS is attached.

Table D.1.3.2-7 lists the 12 EOS caution and warning functions which have been identified by GAC for EOS. Three caution and warning functions vary for alternate Orbit transfer subsystems which vary as a function of the launch vehicle configuration.

TABLE D.1.3.2-7

EOS CAUTION & WARNING FUNCTIONS

SUBSYSTEM	MEASUREMENT	TYPE	CONFIGURATION		
			TITAN III B	WEIGHT CONSTRAINED TITAN III B	D2910
Orbit Adjust	Hydrazine Tank #1 Pressure	Caution	X	X	X
" "	" " #2 "	"	X	X	X
Orbit Transfer	SRM Safe & Arm Device Number 1	Warning	X	X	X
" "	" " " " " Number 2	"		X	X
" "	" " " " " Number 3	"		X	X
Orbit Transfer	SRM Safe & Arm Device Number 4	"		X	X
Electrical Power	Solar Array Safe & Arm Device #1	Warning	X	X	X
" "	" " " " " " #2	"	X	X	X
" "	" " " " " " #3	"	X	X	X
" "	" " " " " " #4	"	X	X	X
" "	" " " " " " #5	"	X	X	X
" "	" " " " " " #6	"	X	X	X

D.1.3.2.3.2 Data Handling Alternate Configurations

The objective of this study is to obtain accurate costing for a data bus system suitable for EOS application. Data Bus system configuration alternatives are many. These include full duplex versus half duplex, separate command and address line versus common lines, data rates, formats, combined versus separate remotes, etc. In order to limit the scope of the study to produce a useful output which would yield accurate costing a baseline data bus system for EOS was selected. Alternative configurations within the baseline were defined and vendors requested to quote on the baseline and these alternatives. Vendors were also invited to quote on their own alternative configuration providing it met the overall operational parameters of the baseline system.

The NASA Standard Full Duplex System with commands and addresses sharing a common bus was selected and merged with the NASA EOS baseline equipment characteristics. The decision to incorporate the NASA Standard data bus features into the baseline system was made for the following reasons and assumptions. The non recurring development costs for such a system are not chargeable (assumption) to the EOS program, thereby, significantly reducing a major cost element of the EOS Data Handling Group. Also a review of its features showed a strong resemblance to the present system being developed for the J.S.C. Orbiter. Orbiter commonality would also reduce program costs. In addition the NASA standard operating at 1 MBPS rate and using self synching manchester II Bi Phase L code easily fulfills EOS requirements.

Since the full duplex system uses a common bus for both commands and addresses it was determined that a single central unit (Controller/Formatter) which would control the bus issuing both addresses and commands would reduce system complexity. The alternative is to operate the bus under control of two separate and distinct units.

The selected baseline is shown in Figure D.1.3.2.-9. Configuration Alternatives to the system are Configuration #1 which uses a remote unit that incorporates both a remote decoder and remote multiplexer (Mux). Configuration 1P is the same remote unit but power strobed with 16 KHz square wave. Configuration 2 uses separate remote decoders and remote mux's while 2P is the 16 KHz square wave power strobed version of #2 (the NASA EOS baseline for remotes).

These alternatives were selected to assess the cost impact of 16 KHz square wave power strobing of remote units and to determine whether combining the remote decoder and remote mux into single remote units offers cost, weight and power savings.

Preliminary conclusions are drawn from three companies, Harris (Radiation), SCI and Spacetac who have responded to date. Prices, weight and powers of individual units have been summed to determine the total cost weight and power, for a typical EOS data bus system consisting of a Bus controller/formatter and eight remote units. These results are plotted in figures D.1.3.2-10, -11 and 12.

The Harris 1P configuration is the lowest cost system for a program where two or more spacecraft are procured (Figure D.1.3.2-10). An eight

EOS DATA BUS SYSTEM CONFIG'S

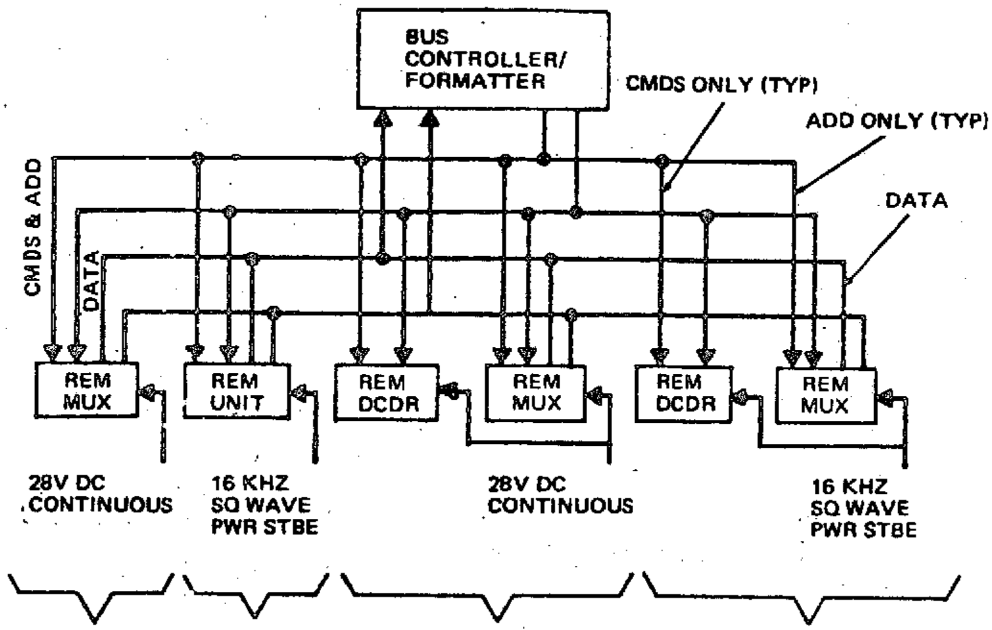


FIG D.1.3.2-9

TOTAL PROGRAM COST (NON RECURRING + RECURRING) OF CANDIDATE DATA BUS SYSTEMS

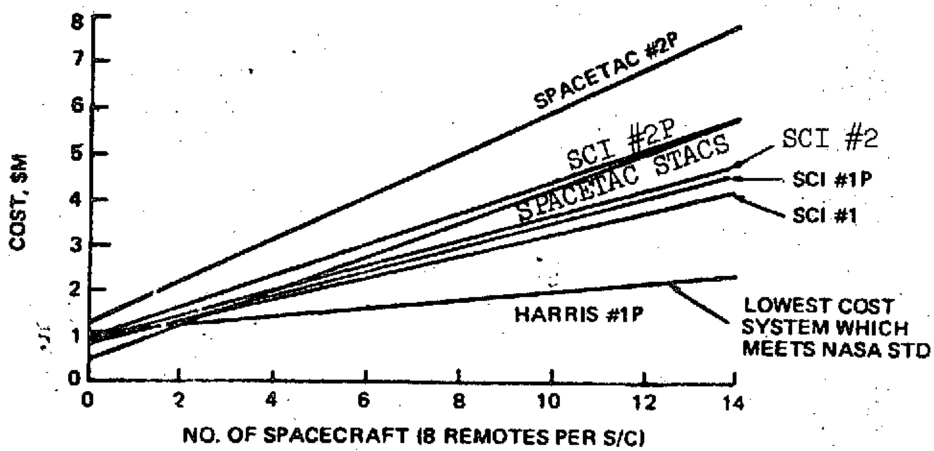


FIG. D.1.3.2-10

remote unit single thread Harris 1P system for five shipsets is 623 K dollars less than the next lowest cost system the SCI #1P and 1.2M dollars less costly for ten spacecraft. The Harris 1P system weighing only 36 pounds is also the lightest weight system (Figure D.1.3.2-11) weighing 19 pounds less than the next lightest weight systems, the SCI 1 and 1P.

The Spacetacs 2P and STACS systems draw the least power 5 watts for a system of 8 remotes, but Harris is the next lowest power consumer, drawing 12 watts per system. Seven watts equates to 7K dollars, a savings which is offset for two or more shipsets by Harris's very low recurring cost of 97K dollars per system.

This combination of relatively low system power combined with very low recurring costs plus lowest weight make the Harris 1P the most attractive candidate for EOS. However, if only one shipset is to be ordered then the Spacetacs STACS configuration is the more cost effective candidate due to its very low non-recurring cost of 464K dollars.

Figures D.1.3.2-10 and -11 also show that combining remotes reduces system cost and weight. The three lowest cost and lightest weight systems use combined remotes. Weight savings is due to extra primary case structure, casing and power regulator electronics required by the separate unit.

The most cost effective systems (Harris 1P and Spactacs STACS) utilize power strobing, indicating that power strobing is cost effective for EOS. However, an analysis of SCI's response, (the only manufacturer to respond to all four alternates) indicates that power strobing its design may not be cost effective. The SCI 1P system draws 22 watts less than its non-power strobed version (22K dollars savings for EOS) but costs 24K dollars more in recurring costs plus 28K dollars additional non-recurring fees. The cost for power strobing is even greater for the separate remotes.

Assuming the Harris 1P system becomes the NASA standard and its one million dollar non-recurring cost is not chargeable to the EOS program, then this system would be the prime choice regardless of number of shipsets ordered due to its very low recurring unit cost. Results of this study are not conclusive since not all of the solicited vendors (including the orbiter prime data bus contractor) have responded to date.

WEIGHT OF CANDIDATE DATA BUS SYSTEMS

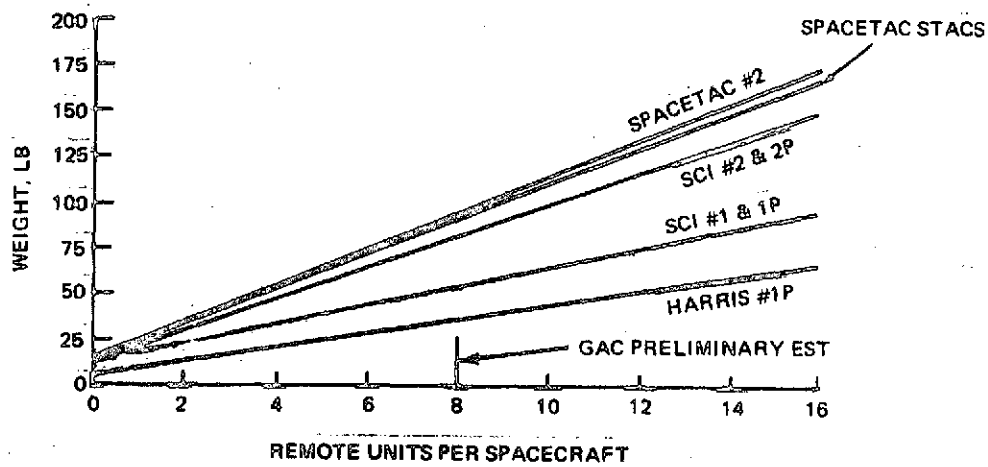


Fig. D.1.3.2-11

COMPUTER MEMORY TYPE TOTAL PROGRAM COST INCLUDING
COST OF POWER (I WATT= IK)

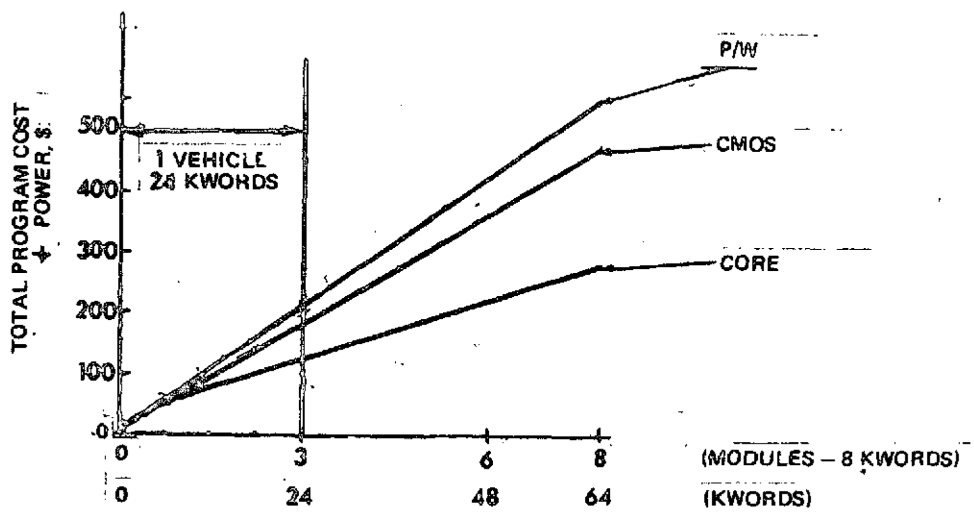
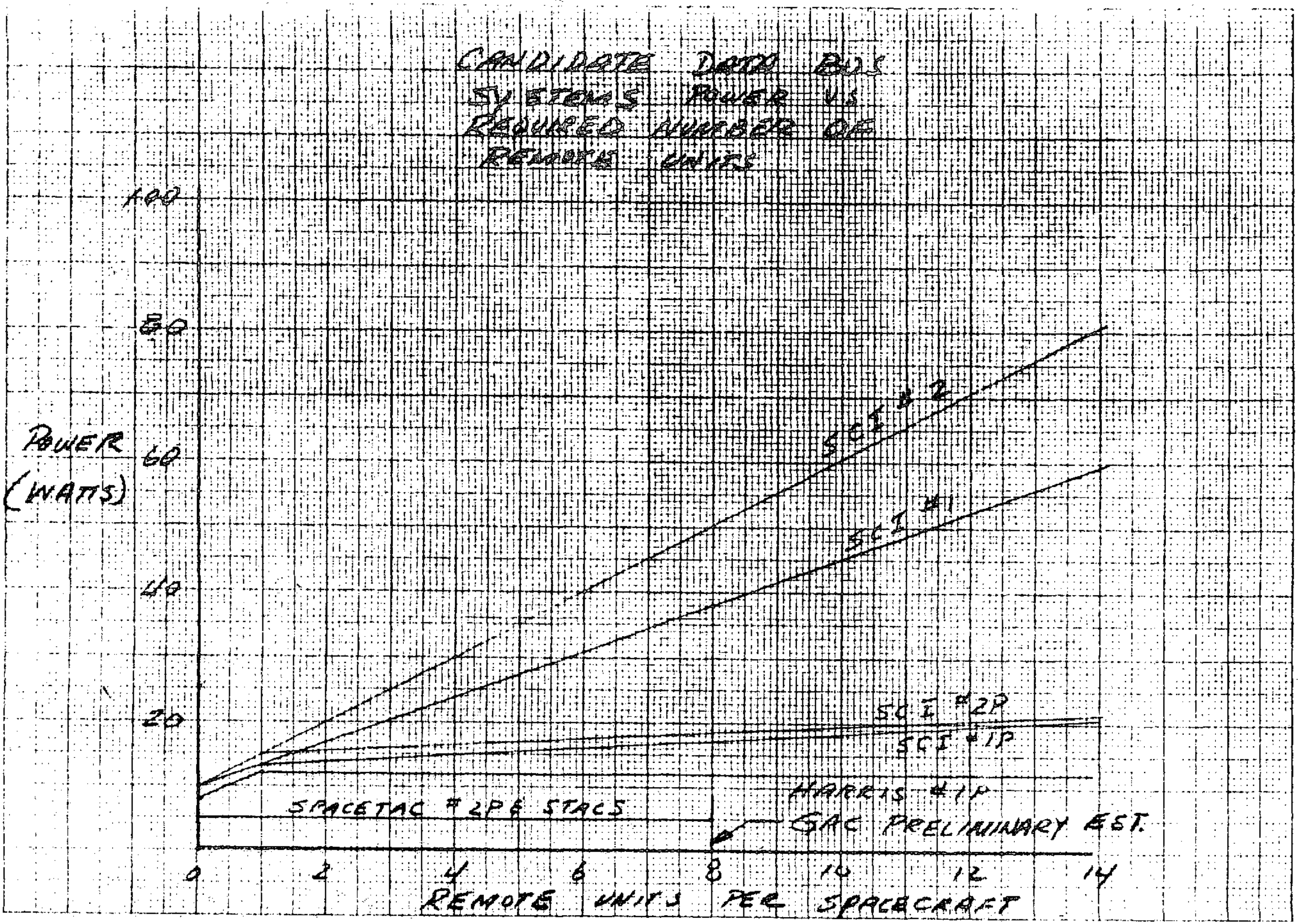


Fig. D.1.3.2-13

FIGURE D.1.3.2-12



D.1.3.2.3.3 DHG AOP Memory Alternatives

The Advanced On-Board Processor (AOP) is available with three memory types: core, plated wire and CMOS (Complimentary Metal Oxide Semiconductor). Core and plated wire are both considered to be acceptable memory types for EOS application while CMOS is conditionally acceptable.

CMOS RAM's (Random Access Memories) are volatile requiring quiescent power at all times to retain their stored data. A power interrupt or loss (e.g. due to high peak load transients including fault loads, clearing shorts, power transfer from orbiter to EOS, battery explosion, etc.) could cause loss of all data in the RAM. If a spacecraft design without an analog backup is selected then it may be desirable to design the computer with its own backup battery which would provide power during any spacecraft power interrupts or shutdown. Another alternative is to store the entire contents of the RAM on an on-board tape recorder and following a shutdown, a routine stored in a CMOS ROM (Read Only Memory) module could direct the recorder to reload the main memory.

The primary driver for memory selection was total program cost. Assuming the value of one watt of spacecraft power is 1K dollars, a plot of total program cost (including power costs) was made (see Figure D.1.3.2-13) As shown, selection of core memory for a single spacecraft requiring 24K memory saves 91K dollars over plated wire and 57K dollars over CMOS.

D.1.3.2.3.4 Data Handling Selected Configuration

The baseline single thread DHG as depicted in Figure D.1.3.2-14 is comprised of a 24K word Advanced Onboard Processor (AOP) with core memory, command decoder, bus controller/formatter unit, seven remote units (one located in the C&DH module, the remaining six distributed throughout the spacecraft), a 4.096 MHZ central clock and signal conditioning units which condition Hi-and-Lo level signals to 0-5VDC, and also contain D/A (Digital to Analog) conversion and latching relays for implementation of commands.

The AOP computer using the Harris CMMA chips will be flown aboard ERTS B. A space qualified AOP minimizes non-recurring costs. Assuming AOP procurement efforts progress as planned, the AOP should be well proven prior to the first EOS flight, thereby minimizing program risk.

Using a standard Aerospace instruction mix of 80% shorts (adds) and 20% longs (multiplies) the AOP's throughput is computed to be 85KOPS, seven to eight times the current maximum requirement for EOS.

The AOP's capability to perform data compression is utilized on housekeeping data, thereby obviating the need for the optional tape recorder. This represents a savings of approximately 80K dollars, 8 watts and 14 pound per spacecraft. The AOP computes and/or stores each measurements high, low, mean, mean variance and current value. Utilizing this technique, 150 measurements require 750 storage locations in main memory. Technique is executed via the 30 words software routine flow charted in Figure 6.7-2.

The selected Harris full duplex data bus system #1P configuration has combined remote units which are power strobed with either 16 KHZ Square Wave or 28 VDC. Remote units have dual receivers and transmitters which operate off the dual redundant command/address busses and data reply busses respectively. Each unit has 64 input channels that can be used for analog, bilevel or serial digital signals as defined in the NASA EOS C&DH Specification. Each unit also has 64 output channels for pulse commands plus 4 serial magnitude command outputs. Output levels are also as defined in the NASA C&DH Specification. Remote units weigh 4 pounds each and draw 4 watts of power when "ON."

The controller/formatter also has dual receivers and transmitters which interface to the dual redundant busses. This unit can accept and interleave 50 commands/second from the command decoder with 62.5 commands/second from the AOP and transmit these to the remote units. Telemetry output rates are command selectable at 32/16/8/4/2/1 KBPS and format consists of minor frames of 128 eight bit words.

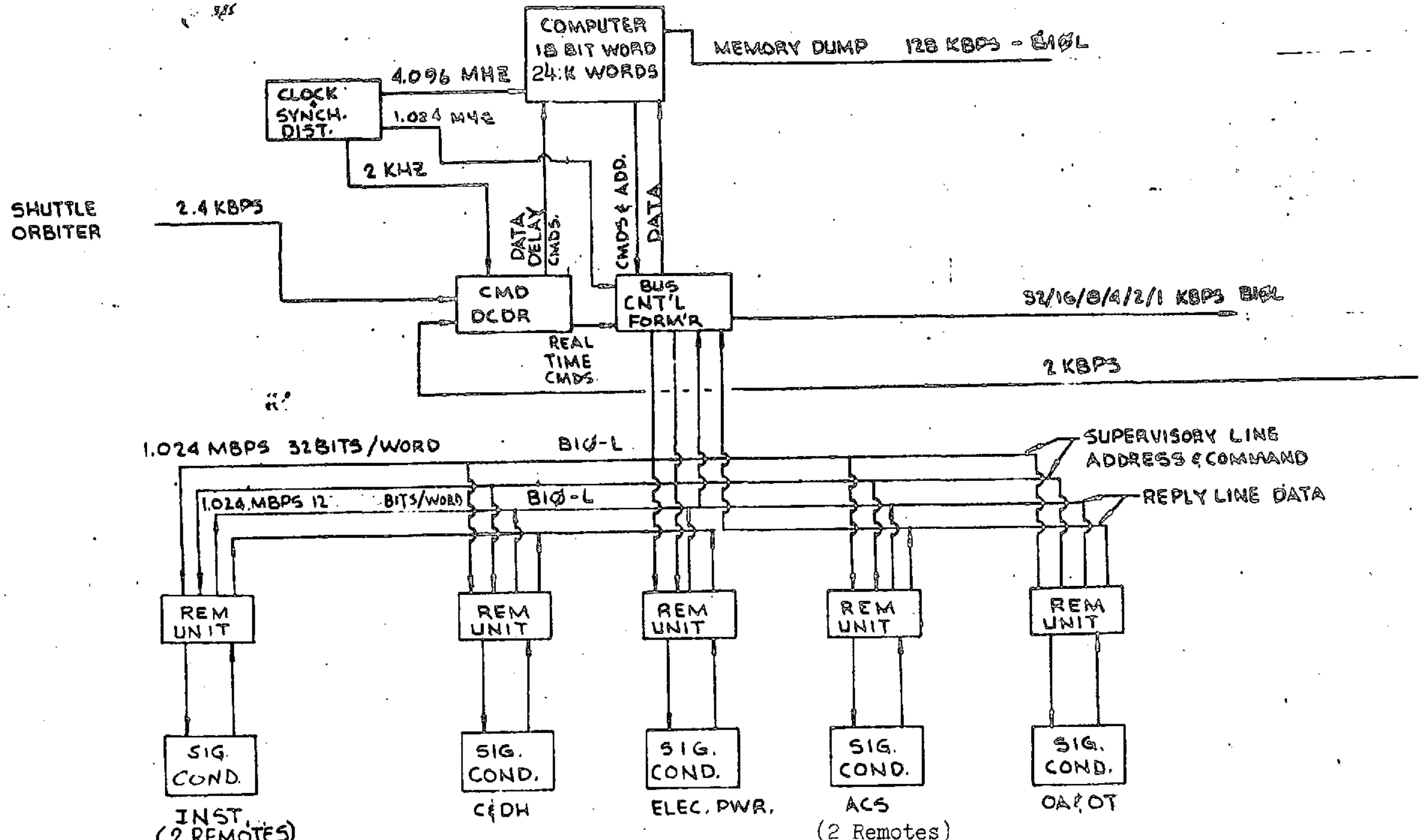


Fig. D.1.3.2-14 Data Handling Group

1.3.2-38

PRELIMINARY

ENCLOSURE D1.3.2.3-1

EARTH OBSERVATORY SATELLITE (EOS)

MEASUREMENTS LIST

JUNE 1974

PREPARED BY: ____________________
T.Newman

APPROVED BY: ____________________
A.Gartenberg

KEY TO MEASUREMENT LIST SYMBOLS

1) Measurement Identification Number

   Format XXYYYZ

   XX = Subsystem

AC	Attitude Control
CD	Communications and Data Handling
EP	Electrical Power
GN	Guidance & Navigation
IN	Instruments
OA	Orbit Adjust
OT	Orbit Transfer
TH	Thermal Control
ST	Structural

   YYY = Number in order by subsystem

   Z = Type of Measurement

C	Current
D	Discrete
F	Frequency
P	Pressure
Q	Quantity
T	Temperature
V	Voltage
W	Word

2) Sample rate give in samples/second

3) Measurement Class

H	Hardline to GSE Connector
T	Telemetry
O	On-board

4) Launch Vehicle Configuration

TIIIB	Titan IIIB
WCT	Weight Constrained Titan
DB2910	Delta 2910

A C S MODULE

PRELIMINARY

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACC-URACY	SAMPLE RATE	MEAS. CLASS H	MEAS. CLASS T	MEAS. CLASS O
AC001 D	Dig. Sun Sens A (16 BPW)		deg.		1		X	X
AC002 D	Dig. Sun Sens B (16 BPW)		deg.		1		X	X
AC003 D	Dig. Sun Sens POWER	ON/OFF			1		X	X
AC004 D	Dig. Sun Sens SUN PRESENCE	PRES/NO PRES.			1		X	X
AC005 A	Dig. Sun Sens Coarse Sig Err. Pitch	±40	deg.	5%	1		X	X
AC006 A	Dig. Sun Sens Coarse Sig Err. Roll	±40	deg.	5%	1		X	X
AC007 A	Dig. Sun Sens. Error Sig. Pitch	±32	deg.	5%	1		X	X
AC008 A	Dig. Sun Sens Error Sig. Roll	±32	deg.	5%	1		X	X
AC009 D	Fix HD. TKR. Power #1	ON/OFF	N/A		1		X	X
AC010 D	Fix HD. TKR. Power #2	ON/OFF	N/A		1		X	X
AC011 A	Fix HD. TKR. Error Pitch	±8	deg.	5%	1		X	X
AC012 A	Fix HD. TKR. Error Yaw	±8	deg.	5%	1		X	X
AC013 V	Fix HD. TKR. Hi Volt	0-1500	Volts	5%	1		X	X
AC014 T	Fix HD. TKR. Int. Temp	0-130	°F	5%	1		X	X
AC015 M	Fix HD. TKR Star Mag.	-1 to +5	Mag.	5%	1		X	X

PRELIMINARY

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACCURACY	SAMPLE RATE	MEAS. CLASS H	T	O
AC016 D	Fine WHL S/C Direct (Pitch)	CCW/CW			1		X	X
AC017 D	Fine WHL S/C Direct (Roll)	CCW/CW			1		X	X
AC018 D	Fine WHL S/C Direct (Yaw)	CCW/CW			1		X	X
AC019 D	Fine WHL Switch to Dig. Sun Sens.	ENG/DIS			1		X	X
AC020 D	Fine WHL Switch to Dig. Sun Sens	ENG/DIS			1		X	X
AC021 T	Fine WHL Int Temp (Pitch)	0-130	^{o}F	5%	1		X	X
AC022 T	Fine WHL Int Temp (Roll)	0-130	^{o}F	5%	1		X	X
AC023 T	Fine WHL Int. Temp (Yaw)	0-130	^{o}F	5%	1		X	X
AC024 P	Fine WHL Press. (Pitch)	0-5	Torr	5%	1		X	X
AC025 P	Fine WHL Press (Roll)	0-5	Torr	5%	1		X	X
AC026 P	Fine WHL Press (Yaw)	0-5	Torr	5%	1		X	X
AC027 R	Fine WHL Speed (Pitch)	0-1000	RPM	5%	1		X	X
AC028 R	Fine WHL Speed (Roll)	0-1000	RPM	5%	1		X	X
AC029 R	Fine WHL Speed (Yaw)	0-1000	RPM	5%	1		X	X
AC030 D	Very High Thrust Jet (VHTJ)(+) Pitch	ON/OFF	N/A		1		X	X
AC031 D	VHTJ (-) Pitch	ON/OFF	N/A		1		X	X
AC032 D	VHTJ (+) Yaw	ON/OFF	N/A		1		X	X
AC033 D	VHTJ (-) Yaw	ON/OFF	N/A		1		X	X
AC034 D	High Thrust Jet (HTJ)(+) Pitch	ON/OFF	N/A		1		X	X
AC035 D	HTJ (-) Pitch	ON/OFF	N/A		1		X	X
AC036 D	HTJ (+) Roll	ON/OFF	N/A		1		X	X
AC037 D	HTJ (-) Roll	ON/OFF	N/A		1		X	X
AC038 D	HTJ (+) Yaw	ON/OFF	N/A		1		X	X
AC039 D	HTJ (-) Yaw	ON/OFF	N/A		1		X	X
AC040D	RGA Pitch Rate (16 BPW)				1		X	X
AC 041D	RGA Roll Rate (16 BPW)				1		X	X
AC042D	RGA Yaw Rate (16 BPW)				1		X	X
AC043D	RGA ATT Null (Pitch)	Yes/No			1		X	X

PRELIMINARY

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACCURACY	SAMPLE RATE	MEAS. CLASS H	T	O
AC 044 D	RGA ATT Null (Roll)	Yes/No			1		X	X
AC 045D	RGA ATT Null (Yaw)	Yes/No			1		X	X
AC 046 D	RGA HTRS.	ON/OFF			1		X	X
AC 047 D	RGA Gyro WHL Speed (Pitch)	Yes/No			1		X	X
AC 048 D	RGA Gyro WHL Speed (Roll)	Yes/No			1		X	X
AC 049 D	RGA Gyro WHL Speed (Yaw)	Yes/No			1		X	X
AC 050 D	RGA Null (Pitch)				1		X	X
AC 051 D	RGA Null (Roll)				1		X	X
AC 052D	RGA Null (Yaw)				1		X	X
AC 053 A	RGA Err. (Pitch)	±40	Sec	1%	1		X	X
AC 054 A	RGA Err. (Roll)	±40	Sec	1%	1		X	X
AC 055 A	RGA Err. (Yaw)	±40	Sec.	1%	1		X	X
AC 056 T	RGA Elec. HS Temp	0-130	°F	5%	1		X	X
AC 057 T	RGA Gyro HS Temp	0-130	°F	5%	1		X	X
AC 058 T	RGA Gyro HS Temp	0-150	°F	5%	1		X	X
AC 059 T	RGA Gyro HS Temp	0-150	°F	5%	1		X	X
AC 060 T	RGA Gyro HS Temp	0-150	°F	5%	1		X	X
AC 061 R	RGA Rate Err. (Pitch)	215-217	Sec/Sec	1%	1		X	X
AC 062 R	RGA Rate Err. (Roll)	±1.0	Sec/Sec	1%	1		X	X
AC 063 R	RGA Err. (Yaw)	±1.0	Sec/Sec	1%	1		X	X
AC 064 D	Low Thrust Jet (LTJ) (+ Pitch)	ON/OFF	-	-	1		X	X
AC 065D	LTJ (- Pitch)	ON/OFF			1		X	X
AC 066D	LTJ (+ Roll)	ON/OFF			1		X	X

PRELIMINARY

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACC-URACY	SAMPLE RATE	MEAS. CLASS H	T	O
AC067D	LTJ (-Roll)	ON/OFF			1		X	X
AC068D	LTJ (+ Yaw)	ON/OFF			1		X	X
AC069 D	LTJ (- Yaw)	ON/OFF			1		X	X
AC070 D	Mag. UNLDG Sys (MUS) Torquer Pwr.	ON/OFF			1		X	X
AC071 G	MUS MagneTometer (Pitch)	0±0.6	GAUSS	5%	1		X	X
AC072 G	MUS MagneTometer (Roll)	0±0.6		5%	1		X	X
AC073 G	MUS MagneTometer (Yaw)	0±0.6		5%	1		X	X
AC074V	MUS Unldg. Coil (Pitch)	0-5	Volts		1		X	X
AC075V	MUS Unldg Coil (Roll)	0-5	Volts		1		X	X
AC076V	MUS Unldg Coil (Yaw)	0-5	Volts		1		X	X
AC077D	Gyro Select (Pitch)				1		X	X
AC078D	Gyro Select (Roll)				1		X	X
AC079D	Gyro Select (Yaw)				1		X	X
AC080D	Remote Unit A Fail				1		X	X
AC081D	Remote Unit B Fail				1		X	X
AC082D	Sig. Cond. Fail	0-5	VDC	±.5%	1		X	X
AC083V	Sig. Cond. (A/D) Cal. 1	0-5	VDC	±.5%	1		X	X
AC084V	Sig. Cond. (A/D) Cal. 2	0-5	VDC	±.5%	1		X	X
AC085V	Sig. Cond. (A/D) Cal. 3	0-5	VDC	±.5%	1		X	X
AC086T	ACS Mod Temp #1	0-120	°F		1		X	X
AC087T	ACS Mod Temp #2	0-120	°F		1		X	X

C & D H MODULE

EOS MEASUREMENTS

PRELIMINARY

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACCURACY	SAMPLE RATE	MEAS. CLASS H	T	O
CD001 D	Computer Fail				1		X	X
CD002 D	Computer Decision Reg. BIT				N/A		X	X
CD003 D	Computer Carry Reg. BIT				N/A		X	X
CD004 D	Computer Overflow Reg. BIT				N/A		X	X
CD005 W	Computer Inst. Save Reg. BIT (11 bits)				N/A		X	X
CD006 W	Computer Inst Add Reg. BIT (18 bits)				N/A		X	X
CD007 W	Computer Mem'y Oprnd Reg. BIT (18 bits)				N/A		X	X
CD008 W	Computer Page Reg. BIT (4 bits)				N/A		X	X
CD009 W	Computer Accumulator Reg. BIT (18 bits)				N/A		X	X
CD010 W	Computer Inst Cntr BIT				N/A		X	X
CD011 W	Computer Extended Acc. BIT				N/A		X	X
CD012 W	Computer Index Reg BIT				N/A		X	X
CD013 W	Computer DMA Blck Lngth Add (18 bits)				N/A		X	X
CD014 W	Computer DMA Cycle Stl Add (18 bits)				N/A		X	X
CD015 D	CMD DCDR Stdby	Off/On			50		X	X
CD016 D	CMD DCDR Operate	Off/On			50		X	X
CD017 D	CMD DCDR Execute	Off/On			50		X	X
CD018 D	CMD DCDR Reject	Off/On			50		X	X
CD019 W	Veh. Time Code Word #1	30 par bits	m sec	10%	50		X	X
CD020 W	Veh. Time Code Word #2	30 par bits	m sec	10%	50		X	X
CD021 D	Bus Cont'lr Form Fail				1		X	X
CD022 D	Remote Unit Fail				1		X	X
CD023 V	Receiver Temperature	0 to 5	VDC	5 %	1		X	X
CD024 V	RDU Decoder Activate	0 to 20	VDC	5 %	1		X	X

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACCURACY	SAMPLE RATE	MEAS. CLASS H	T	O
CD025 V	RDU Carrier Presence	0 to 20	VDC	5%	1		X	X
CD026 V	RDU State Phase Error	0 to 5	VDC	5%	1		X	X
CD027 V	RDU Signal Strength	0, + 20	VDC	5%	1		X	X
CD028 V	XMTR RF Power	0 to 5	VDC	5%	1		X	X
CD029 V	XMTR Temperature	0 to 5	VDC	5%	1		X	X
CD030 D	Ranging On	0, +20	VDC	5%	1		X	X
CD031 D	Antenna Select Position	0, +20	VDC	5%	1		X	X
CD032 D	Sig. Cond. Fail	0-5	VDC	±.5%	1		X	X
CD033 V	Sig. Cond (A/D) Cal. 1	0-5	VDC	±.5%	1		X	X
CD034 V	Sig. Cond (A/D) Cal. 2	0-5	VDC	±.5%	1		X	X
CD035 V	Sig. Cond (A/D) Cal. 3	0-5	VDC	±.5%	1		X	X
CD036 T	C & DH Mod Temp #1	0-120	°F		1		X	X
CD037 T	C & DH Mod Temp #2	0-120	°F		1		X	X
CD038 T	Computer Temp	0-120	°F		1		X	X
CD039 T	Controller/Form Temp	0-120	°F		1		X	X

ELEC. POWER MODULE

EOS MEASUREMENTS

PRELIMINARY

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACC- URACY	SAMPLE RATE	MEAS. CLASS H	T	O
EP-001T	Solar array temp A	-100 to+200	Deg F	± 2°F	1		X	X
EP-002T	Solar array temp B	-100 to+200	Deg F	± 2°F	1		X	X
EP-003T	Solar array temp C	-100 to+200	Deg F	± 2°F	1		X	X
EP-004T	Solar array temp D	-100 to+200	Deg F	± 2°F	1		X	X
EP-005C	Solar array current	0 to 20	AMP	± 0.2A	1		X	X
EP-006T	Battery 1 temp	0 to 140	Deg F	± 2°F	1		X	X
EP-007C	Battery 1 Hi current	-20 to +30	AMP	±0.2A	1		X	X
EP-008V	Battery 1 voltage	20 to 33	Volt	± 0.1V	1		X	X
EP-009D	Bat 1 volt/temp sense	Disab/Enabl	-	-	1		X	X
EP-010D	Bat 1 HTR Status	Off/On	-	-	1		X	X
EP-011D	Bat 1 Status	Disab/Enab	-	-	1		X	X
EP-012T	Battery 2 temp	0 to 140	Deg F	± 2°F	1		X	X
EP-013C	Battery 2 Hi current	-20 to +30	AMP	± 0.2A	1		X	X
EP-014V	Battery 2 voltage	20 to 33	Volt	± 0.1V	1		X	X
EP-015D	Battery 2 volt/temp sense	Disab/Enab	-	-	1		X	X
EP-016D	Bat 2 HTR status	Off/On	-	-	1		X	X
EP-017D	Bat 2 status	Disab/Enab	-	-	1		X	X
EP-018V	Bat chrgr input volt	40 to 125	Volt	±1V	1		X	X
EP-019T	Bat chrgr temp	0 to 140	Deg F	± 5°F	1		X	X
EP-020D	Bat volt LIMIT level 1	Off/On	-	-	1		X	X
EP-021D	Bat volt limit level 2	Off/On	-	-	1		X	X
EP-022D	Bat volt limit level 3	Off/On	-	-	1		X	X
EP-023D	Bat volt limit level 4	Off/ON	-	-	1		X	X
EP-024D	Bat chrgr float lvl	Disab/Enab	-	-	1		X	X
EP-025D	Bat Charge trickle	Disab/Enab	-	-	1		X	X
EP-026V	PDU bus voltage	20 to 33	Volt	± 0.1V	1		X	X

EOS MEASUREMENTS

PRELIMINARY

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO HI	ENG. UNITS	ACC-URACY	SAMPLE RATE	MEAS. CLASS H	T	O
EP-027C	Total bus currents	0 to 20	AMP	±0.2A	1		X	X
EP-028C	ACS mod. bus current	0 to 20	AMP	±0.2A	1		X	X
EP-029C	C & DH mod. bus current	0 to 20	AMP	±0.2A	1		X	X
EP-030C	OA & T mod. bus current	0 to 20	AMP	±0.2A	1		X	X
EP-031C	Inst, mod bus current	0 to 20	AMP	±0.2A	1		X	X
EP-032A	SAD Wide Angle Sun Sensor	180 to+180	Deg	± 5%	1		X	X
EP-033T	SAD motor temp	0 to 160	Deg F	± 5%	1		X	X
EP-034A	SAD shaft position	0 to 360	Deg	± 5%	1		X	X
EP-035D	EED Prim bus status	Off/On	-	-	1		X	
EP-036D	EED SEC bus status	Off/On	-	-	1		X	
EP-037D	SLR ARR launch INH status	Off/On	-	-	1		X	X
EP-038D	Remote Unit Fail				1		X	X
EP-039D	Sig. Cond. Fail	0-5	VDC	±.5%	1		X	X
EP-040V	Sig. Cond (A/D) Cal. 1	0-5	VDC	±.5%	1		X	X
EP-041V	Sig. Cond (A/D) Cal. 2	0-5	VDC	±.5%	1		X	X
EP-042V	Sig. Cond (A/D) Cal. 3	0-5	VDC	±.5%	1		X	X
EP-043T	Elec. Pwr Mod Temp #1	0-120	°F		1		X	X
EP-044T	Elec. Pwr Mod Temp #2	0-120	°F		1		X	X

ORBIT ADJUST AND TRANSFER (OA & T)

MODULE

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO-HI	ENG. UNITS	SAMPLE RATE	MEAS. CLASS H	T	O	LAUNCH VEH. CONGIG. T IIIB	WCT	2910
OA001P	Hydrazine Tank #1 Press.			1	X	X		X	X	X
OA002P	" " #2 "			1	X	X		X	X	X
OA003T	" " #1 Temp.			1		X		X	X	X
OA004T	" " #2 Temp.			1		X		X	X	X
OA005T	Thruster Ass'y. #1 Temp			1		X		X	X	X
OA006T	" " #2 "			1		X		X	X	X
OA007T	" " #3 "			1		X		X	X	X
OA008T	" " #4 "			1		X		X	X	X
OA009T	" " #5 "			1		X		X	X	X
OA010T	" " #6 "			1		X		X	X	X
OA011T	" " #7 "			1		X		X	X	X
OA012T	" " #8 "			1		X		X	X	X
OA013T	" " #9 "			1		X		X	X	X
OA014T	" " #10 "			1		X		X	X	X
OA015T	" " #11 "			1		X		X	X	X
OA016T	" " #12 "			1		X		X	X	X
OA017T	" " #13 "			1		X		X	X	X
OA018T	" " #14 "			1		X		X	X	X
OA019T	" " #15 "			1		X		X	X	X
OA020T	" " #16 "									

PRELIMINARY

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO-HI	ENG. UNITS	SAMPLE RATE	MEAS. CLASS			LAUNCH VEH. CONGIG.		
					H	T	O	T IIIB	WCT	2910
OA021T	Thruster Ass'y. # 17 Temp.			1		X		X	X	X
OA022T	" " # 18 "			1		X		X	X	X
OA023T	" " #19 "			1		X		X	X	X
OA024T	" " #20 "			1		X		X	X	X
OA025D	Latching Valve #1 O/C			50		X				
OA026D	" " #2 O/C			50		X				
OA027D	" " #3 O/C			50		X				
OA028D	Remote Unit Fail			1		X	X	X	X	X
OA029D	Sig. Cond. Fail	0-5	VDC	1		X	X	X	X	X
OA030V	Sig. Cond (A/D) Cal. 1	0-5	VDC	1		X	X	X	X	X
OA031V	Sig. Cond (A/D) Cal. 2	0-5	VDC	1		X	X	X	X	X
OA032V	Sig. Cond (A/D) Cal. 3	0-5	VDC	1		X	X	X	X	X
OA033T	OA & T Mod Temp #1	0-120	°F	1		X	X	X	X	X
OA034T	OA & T Mod Temp #2	0-120	°F	1		X	X	X	X	X
OT001D	SRM Safe & Arm Device Number 1			1		X	X	X	X	X
OT002D	" " " " " 2			1		X	X	X	X	X
OT003D	" " " " " 3			1		X	X	X	X	X
OT004D	" " " " " 4			1		X	X	X	X	X

MEASUREMENTS AND COMMANDS

EXTERNAL TO MODULES

EOS MEASUREMENTS

MEAS. I.D.	MEASUREMENT DESCRIPTION	RANGE LO-HI	ENG. UNITS	SAMPLE RATE	MEAS. CLASS			LAUNCH VEH. CONFIG.				
					H	T	O	TB	TP	DB	DP	2910
TH001T	STRUCTURE TEMP. #1			1		X						
TH002T	" " 2			1		X						
TH003T	" " 3			1		X						
TH004T	" " 4			1		X						
TH005T	" " 5			1		X						
TH006T	" " 6			1		X						
TH007T	" " 7			1		X						
TH008T	" " 8			1		X						
TH009T	" " 9			1		X						
TH010T	" " 10			1		X						
TH011T	" " 11			1		X						
TH012T	" " 12			1		X						
TH013T	" " 13			1		X						
TH014T	" " 14			1		X						
TH015T	" " 15			1		X						
TH016T	" " 16			1		X						
TH017T	" " 17			1		X						
TH018T	" " 18			1		X						
TH019T	" " 19			1		X						
TH020T	" " 20			1		X						

1.3.3 Electrical Power Subsystem

1.3.3.1 - Requirements Definitions

The electrical power subsystem will consist of a standardized power subsystem module and a mission peculiar solar array. The EPS shall satisfy the basic EOS mission and spacecraft requirements as well as being adaptable to a variety of other earth orbiting spacecraft missions.

The power module shall be capable of controlling storing, distributing and monitoring the power derived from the solar array. Energy storage and control functions shall be of a modular design so as to permit optimization of subsystem performance, weight, reliability and cost to specific mission requirements. Command and telemetry requirements shall be compatible with remote command decoding and telemetry multiplexing equipment contained in the power module.

TABLE 1.3.3-1 defines tentative spacecraft/mission electrical power requirements. The basic spacecraft (exclusive of mission peculiar payloads) is estimated to require approximately 300 watts of orbital average power. EOS instruments and other associated payload equipment can range from an average of 150 watts to over 350 watts. Including missions other than EOS could result in an average payload power of up to 500 watts. Therefore, the electrical power subsystem design load capability, based upon this tentative load analysis, should be in the range of 400 to 1000 watts orbital average.

The peak load requirement was defined by NASA in the EPS subsystem specification to be 5.6 KW for 10 minutes, either day or night. This requirement, which is assumed to be contributable to the peak power required by a Synthetic Aperature Radar was reviewed and now appears to be somewhat high. Present estimates of the SAR delta-peak power required are in the order of 1.3 KW over the normal spacecraft load. Therefore, maximum peak loads for the EOS, are not expected to exceed approximately 2 KW.

These power requirements as well as other requirements which have a major impact on the electrical power subsystem design, configuration, performance, weight,

Table 1.3.3-1-A

EOS ELECTRICAL LOADS

BASIC SPACECRAFT

	No. of Units No. Opr.	Avg. Operational Power Range (Watts)	Avg. Operational Power Design (Watts)	Surv. Avg. Pwr. (Watts)
o ACS Module				
– Rate Gyro	4/3	-23-	23	–
– Star Tracker	2/2	-28-	28	–
– Reaction Wheels	3/3	3-45	21	–
– Magnetic Torquers	3/3	8-16	10	10
– Processor Elec.	1/1	-10-	10	10
– Remote Decoder	2/2	Negl.	–	–
– Remode MUX	2/2	Negl.	–	–
ACS Sub-total			92	20
o C&DH Module				
– Transponder		-6-	6	6
– CMD Decoder		-3-	3	3
– Format Gen		-2-	2	2
– Clock		-4-	4	4
– Computer		-47-	47	38
– Tape Recorder		-8-	8	–
– Remote Decoder		Negl.	–	–
– Remote MUX		Negl.	–	–
– Sensors		-10-	10	10
C&DH Sub-total			80	63

GRUMMAN

TRADE STUDY REPORT

TITLE	TRADE STUDY REPORT NO. 1.3.3
	WBS NUMBER 1.7.3.2/1.7.3.7

Table 1.3.3-1A (cont.)

	No. of Units No. Opr.	Avg. Operational Power Range (Watts)	Avg. Operational Power Design (Watts)	Surv. Avg. Pwr. (Watts)
• Power Module				
— Sig. Cond Assy.		-24-	-24-	-24-
— Solar Array Drive		12-20	-20	-20-
PM Sub-total			44	44
• Thermal Control				
— Gyro HTRS*		-10-	10	10
— Thruster HTRS*		-15-	15	15
— Misc. HTRS*		75-125	75	75
T.C. Sub-total			100	100
• Pneumatics/Orbit Adj. Mod.				
— Thrusters		200W/50ms	negl./50ms	Negl.
P/OAM Sub-total				Negl.
			NORM OPS	SURV.
Basic Spacecraft Total			316	227W

*Powers indicated are allocations for preliminary sizing and may not necessarily be required.

PREPARED BY	GROUP NUMBER & NAME	DATE	CHANGE LETTER
			REVISION DATE
APPROVED BY			PAGE 1.3.3-3

*GAC 3711 3-72

TITLE

TRADE STUDY REPORT NO. 1.3.3

WBS NUMBER 1.7.3.2/1.7/3.7

TABLE 1.3.3 -1B

EOS ELECTRICAL LOADS

INSTRUMENT PAYLOADS

Instrument	Min. Cont. Pwr (W)	Delta Pwr (W)	Delta Duty Cycle (%)	Avg Power Range (W)	Avg Power Design (W)
TM	45	0-155	0-50	45-122	45
HRPI	45	0-155	0-50	45-122	45
DCS	10	30	30	-19-	19
SAR	-	500-1500	20	100-300	200
PMMR	-	75-150	20	15-30	20
MSS	65	-	-	-65-	65
W.B. COMM & D.H.	35	350	10	-70-	70
SMM Inst.	174	-	-	-174-	174
SEASAT Inst.	360	280	50	-500-	500
SEDA Inst.	75	70	50	-110-	110

Mission	TM	HRPI	DCS	SAR	PMMR	MSS	W.L. & D.H.	SMM Inst.	SEASAT Inst.	SEOS Inst.	Power: Basic S/C (Watts)	Power: Inst. Pwr	Total Orbital Avg Design Pwr (W)
Delta 2910	X		X			X	X				316	199	515
LRM	X	X	X				X				316	179	495
SMM								X			316	174	490
SEASAT									X		316	500	816
SEOS										X	316	110	426
EOS-B	X		X	X			X				316	334	650

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and cost are summarized in Table 1.3.3-2. The general subsystem design requirements common to all subsystems have not been included in the table.

Table 1.3.3 -2 MAJOR EPS DRIVING REQUIREMENTS

Reqmt/Range			Influence
Mission			
Orbit:	a)	200 to 900 n mi, 0-90° including sun-synch @ 9:30 AM to 12 Noon	Affects dark & light durations and radiation environment which in turn affect EPS sizing, & weight & cost.
	b)	Geosynchronous	
Life:	a)	operational 2 to 5 years	Affects battery life (depth-of-discharge) and solar array degradation - Direct influence on size, weight and cost of EPS
	b)	design 5 years	
Pointing:	a)	earth-pointing	Affects solar array configuration, orientation and size
	b)	inertial pointing	
Spacecraft			
Power:	a)	Orbital average 400 to 1000 watts	Direct influence on size, weight cost of overall EPS
	b)	peak (delta) 1.3 KW, for 10 minutes	
Prime bus Voltage:		28 ± 7 VDC	Affects EPS configuration by eliminating need for prime power conditioning

1.3.3.2. Configuration Alternatives

1.3.3.2.1 - General System Considerations

In evaluating the overall configuration of the prime EPS functions of solar energy conversion, control, energy storage, control and power distribution, specific EOS orbit, life and power requirements were considered only as typical within the more general requirements for true multi-mission capability. Flexibility to efficiently accommodate the full range of requirements was a major criteria in selection of an overall EPS configuration.

The interface between a standardized, multi-capability power module and a mission peculiar solar array was a key consideration in evaluating overall system flexibility. Space craft solar arrays have historically represented approximately 50% of the total EPS cost and any solar array cost savings usually result in reducing the total EPS cost. For some missions, fixed solar arrays and/or existing solar array designs and configurations may be the most cost-effective approach to satisfying the mission requirements provided efficient utilization can be made of available solar array power. The efficient utilization of solar array power is in part, governed by the configuration and components within the power module.

Several basic types of solar array control and energy storage and control configurations were considered for application within the standardized power module.

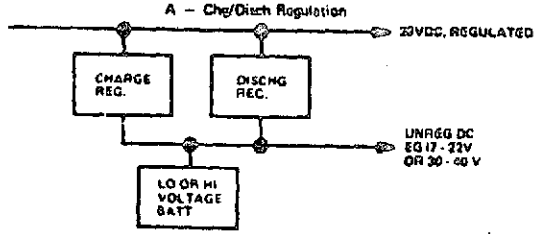
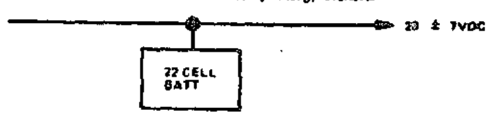
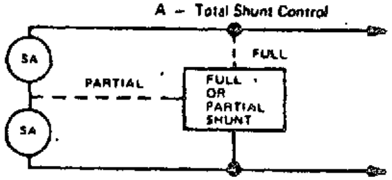
General solar array control techniques that were considered were:

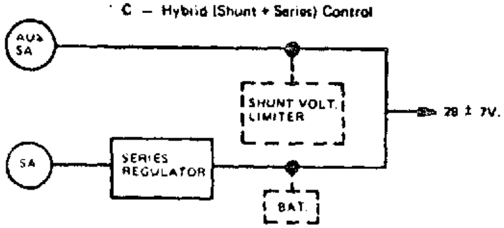
- SHUNT
  - Full or partial, active regulators
  - Switching "on-off" control of array segments
  - Passive voltage limiting - such as "on-array" zener diodes
- SERIES
  - Active, PWM regulation with and without solar array maximum power tracking.
- HYBRID - (both series and shunt control)
  - Combinations of above shunt and series control approaches.

Table 1.33-3 identifies some of the major advantages and disadvantages of the three major types of control. On a quantitative basis, mission peculiar requirements and trade-offs would tend to identify either the series or shunt approach as being optimum for that particular mission, however, for the power system that is standardized yet capable of satisfying many missions, the preferred approach is that with the most flexibility. Within the scope of the types of solar array control techniques, only the hybrid approach is satisfactory.

The problem of efficiently utilizing solar array power where the solar array max. power point is not matched to the prime system voltage is most effectively solved by the use of series regulation. The capability to track the array max. power point allows efficient utilization of thermal transient energy, when this energy is significant. On the other hand, maximum power tracking of the array for some situations may actually reduce the overall system efficiency, in which case, provisions are included to short out the series regulator/MPT for initial battery charging.

Table 1.3.3-3 EPS System Configuration Alternatives

<table>
<tr><th>EPS System Config. Alternatives</th><th>Advantages</th><th>Disadvantages</th></tr>
<tr><td>I. Battery Charge/Discharge Control
A – Chg/Disch Regulation

</td><td>• Prime bus voltage regulation generally better than direct transfer control.
• The series charge regulator need pass only the battery charge power.</td><td>• Discharge regulator increases battery discharge energy required
• Peak loads should operate at different than "standard" bus voltage.
• Source impedance and noise generally higher than with direct transfer.</td></tr>
<tr><td>B – Direct Battery Energy Transfer

</td><td>• Minimum battery energy required
• Minimum source impedance
• Peak loads operate from standard bus voltage</td><td>• Voltage regulation not as good as with chg/dischg regulator.</td></tr>
<tr><td>II. Solar Array Control
A – Total Shunt Control

</td><td>• Maximum power transfer of solar array power output to load and battery.</td><td>• Requires close match of solar array characteristics and system voltage for efficient solar array power utilization.
- Solar array flexibility limited in order to maintain high overall system efficiency.
• Excess solar array power dissipated in S/C</td></tr>
<tr><td>B – Total Series Control

</td><td>• Solar array characteristics / system mismatch less critical
• Can be made to track the max. power capability of solar array.
• Excess solar array power dissipated on array.</td><td>• All S/C power (load + battery charge) passes through series regulator.
• System load capability more directly affected by power handling capability of series regulator.</td></tr>
<tr><td>C – Hybrid (Shunt + Series) Control

</td><td>• Provides maximum flexibility in;
- System load capability
- Solar array design / configuration
- Overall system optimization
• By changing ratio of Aux. SA to Main SA, system can become a total series system (No Aux. SA) to total shunt system with dedicated load & battery recharge solar array segments.</td><td></td></tr>
</table>

Disadvantages of the total series approach are that all spacecraft load and battery charge power passes through the series regulator and is therefore subject to the inefficiency losses of the regulator. In addition, the power handling capability of the regulator limits the max. system load capability. As a remedy for this situation, a dedicated segment of the solar array termed an auxiliary array, is operated in a direct energy transfer mode and supplies its power directly to the spacecraft load without passing through the series regulator. Control of the auxiliary array can be as simple as sizing the array such that it can never supply the entire spacecraft power and is therefore inherently voltage limited by the batteries.

For missions where the auxiliary array degradation in power capability may be significant and/or reduction in spacecraft load can be predicted in advance "on-off" control of auxiliary array circuits (shorting of circuits with solid state switches or relays) could be effectively implemented and controlled by ground and/or spacecraft computer commands. For some missions where overall system efficiency may be optimumized if all spacecraft load power is supplied by the auxiliary array, zener diodes, located on the array may be an effective means of voltage limiting.

For EOS and follow-on missions, the use of a series regulator with capability to be shorted out or max. power track a part or all of the solar array, in conjunction with an auxiliary array will provide the maximum flexibility to accommodate a wide range of mission and spacecraft requirements.

The configuration of the energy storage and regulation functions in the power module are to a large degree selected on the basis of the allowable prime bus voltage of 28 ± 7VDC and the high delta peak load currents required by the Synthetic Aperature Radar. Peak load current battery discharge rates will be minimized (and therefore extend battery life) by using high battery voltages.

On the other hand, use of an extra high voltage that requires a buck regulator to satisfy prime bus voltage requirements, places a restriction on peak loads and/or compromises the regulator design. Since a 22 cell, Nickel-Cadmium battery will satisfy the prime bus voltage requirements without additional active regulation and will provide satisfactory peak load capability, this configuration is selected for the standardized, power module.

1.3.3.2.2 Power Module Alternatives

The basic functions included in the power module are:

- Solar array control
- Energy storage control
- Energy storage
- Interface control
- Command, telemetry and monitoring

The major EPS functional requirements of energy storage and control and solar array control can all be implemented with existing or slightly modified equipment with little or no risk in developing new equipment. A case in point is the demonstration model EPS fabricated by NASA. This baseline system satisfies the basic EOS requirements for a modular, multi-mission capability EPS. Therefore, the major thrust of the Grumman EPS design-cost trades were to identify cost effective improvements to the basic NASA configuration.

Two subsystem functions were considered as likely candidates for improving the cost/performance characteristics of the demonstration power module.

- Solar array battery control
- Battery alternatives

1.3.3.2.2.1 Solar Array/Battery Control Alternatives

As discussed in the preceding general EPS system considerations, the preferred approach to solar array control is the use of series regulation of a portion of the solar array and direct-energy-transfer, battery clamping of the remainder of the solar array. The direct-energy-transfer, auxiliary array is sized and configured to support some percentage of the spacecraft load. The remainder of the load plus the battery recharge power is processed by the series regulator. In the GSFC demonstration power module, this series regulation function is performed by the OAO Power Control and Regulator Units. Figure 1.3.3-1 depicts the system load capability limitations of an OAO system that incorporates 120 A.H. of battery capacity. The figure is based upon a typical EOS orbit of 400 nautical miles. As can be seen in the figure, the OAO PCU and PRU will adequately satisfy the full range of EOS and identified follow-on mission requirements.

As alternatives to the OAO control equipment, consideration was given to two other series regulators that are, at least superficially, suitable for EOS.

- Modified Skylab/ATM Charger-Battery-Regulator Module (CBRM)
- Modified USAF/ELMS Battery Charger

The Skylab/ATM CBRM is a single package that includes a series regulator, a line regulator and a 24 cell, 20 A.H battery. Since the line regulator function is not required for EOS, consideration was given to repackaging just the series regulator section. This was rejected due to the time, cost and risk of repackaging plus the need to modify a major portion of the basic circuitry in the unit.

The other alternative considered was a series regulator/battery charger that was originally designed for the Skylab Orbital Workshop power system and subsequently modified for the Grumman/USAF ELMS program. This basic charger has a number of features which are particularly well suited to a modular EPS that is required to satisfy a wide range of orbital power requirements. A summary comparison of the OAO equipment versus the ELMS battery charger is given in Table 1.3.3-4.

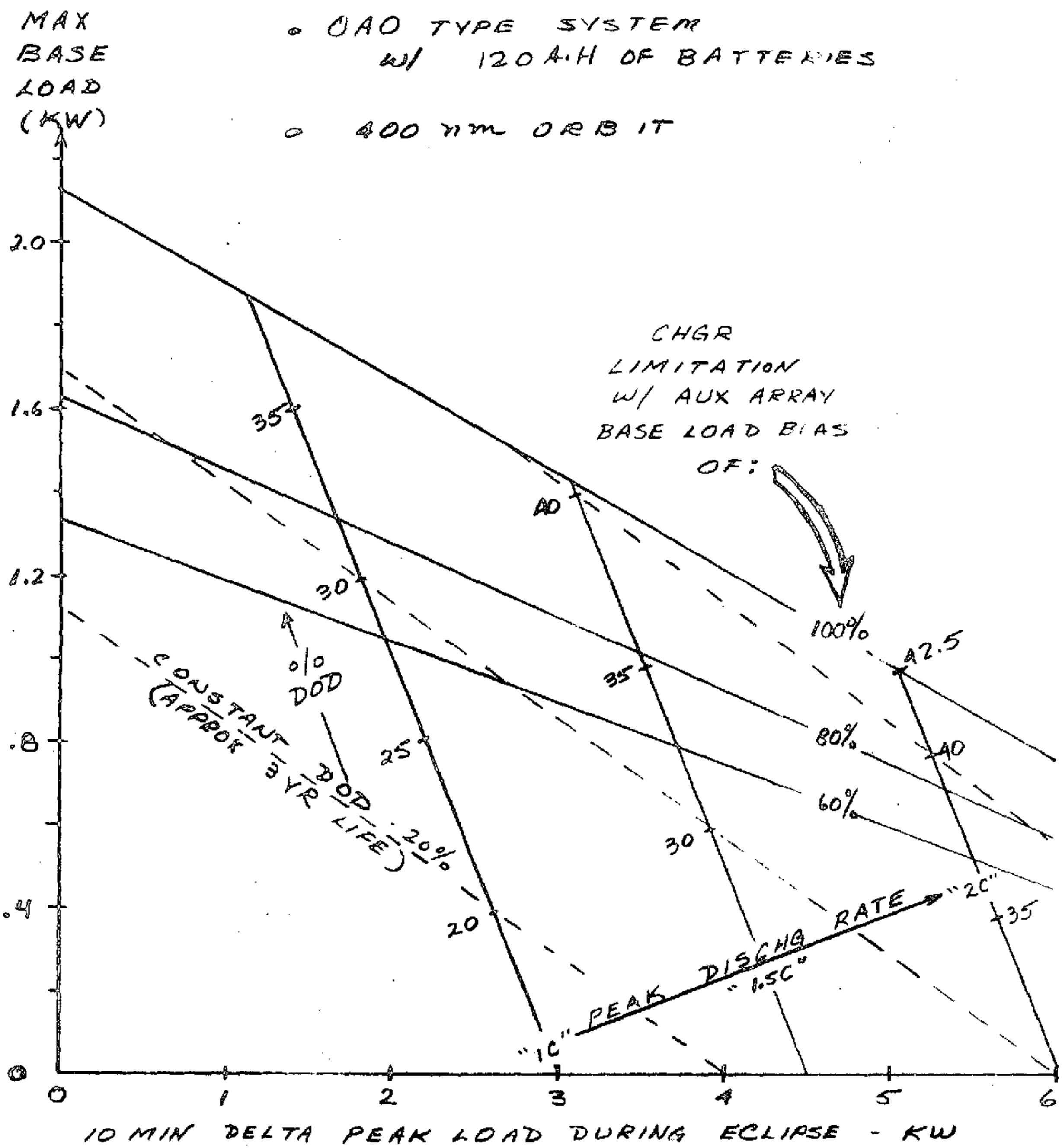


Fig. 1.3.3-1 Electrical Power Subsystem Load Handling (Limitations/Capability)

TITLE	TRADE STUDY REPORT NO. 1.3.3
	WBS NUMBER 1.7.3.2/1.7.3.7

ALTERNATIVE SOLAR ARRAY/BATTERY CONTROL COMPARISON TABLE 1.3.3 - 4

Name	History/Usage	Key Features	Size/Volume	Weight	Non-recurring Cost
Power control Power Regulator Units	Flown on OAO	• Separate regulator package (PRU) and control package (PCU) • PCU also includes misc. battery control & switching functions • PRU & PCU uses standby unit redundancy • Output power capability 2350 watts	each unit 21 5/8 x 10 5/8 x 4 total 1840 in^3	PCU 34.5 lbs PRU 40.5 lbs total 75 lbs	203 K
Modified ELMS Battery charger	1 originally used on Skylab – OWS 2 modified for upcoming flight on USAF-ELMS 3 EOS charger to be modified version of ELMS	• Self contained unit • Modular construction • Redundant Max. Power Trackers (closed loop, accurate to > 97%) • Input voltages from 32 to 125V (lower, voltage limit one of mods to ELMS unit.) • Incorporates up to 5 power regulator modules which, are parallel operated, # of modules can be optimized to maintain high efficiency • Current limited @ 15A/ power module • Power output capability, max 2350 watts	10 x 10 x 7 (700 in^3)	24 lbs	ELMS 56.4K EOS – 65K* *includes percentage of Distribution Unit cost required by charger

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Modifications to the present ELMS battery charger were considered for the EOS;

- Modify drive and control circuits to allow minimum input voltages of 35 VDC.
- Addition of provisions for independent battery charge control sensing for multiple, parallel batteries.
- Incorporation of required charger mode control functions inside charger package.

The volume required for the mode control functions is too large to be placed in the existing package. Since the location of these functions is not critical this modification was eliminated in favor of locating the functions in a central, power control unit.

The original Skylab and present ELMS battery charger minimum allowed solar array max. power voltage is approximately 45 V.D.C. By limiting the minimum voltage to this value versus allowing the minimum voltage to be as low as 33 to 35 volts, charger efficiency is reduced from its optimum by about 2%. Modification of the charger drive and control module circuits can be made which will allow an input voltage minimum of 33-35 VDC. The cost of this change could be offset by the reduction in required solar array area plus provide the additional advantages of reducing power module heat generation and increase the flexibility to accommodate a wider range of solar array characteristics.

One of the general system design considerations that has a direct influence on the series regulator/battery charger is the requirement to charge control a group of batteries with stored energy from approximately 40 to 120 A.H. This stored energy requirement could be satisfied with between two and six 20 A.H batteries, however, with as many as six batteries charged and discharged in parallel, potential problems would occur with load sharing as a result of inter-battery thermal gradients. Providing acceptable temperature differences between the batteries would complicate an already complex thermal design.

As an alternative to the use of large quantities of batteries, consideration was given to satisfying the battery capacity option with two different capacity batteries. The 20 A.H. size battery would be used for total storage requirements of 40 to 60 A.H., while larger capacity batteries of 36 or 45 A.H. would be used when total system capacity requirements are in range of 60 to 120 A.H.

With the use of two different capacity batteries:

- The full range of storage requirements could be satisfied with either 2 or 3 batteries.
- Inter-battery temperature differences could be more easily minimized and therefore improve the battery load sharing and charge control.
- Changes to the battery charger - charger control circuits would not be necessary from mission to mission, and therefore the charger could be of a more "standardized" design.
- Power module wiring and battery control and monitoring functions would not be so mission dependent.
- Power module - non recurring costs due to battery capacity option requirements would be reduced.

In the case of the OAO Power Control Unit, provisions are already included for independent charge control of up to 3 batteries. The ELMS battery charger can presently sense only two batteries, and therefore should be modified to accommodate three batteries. This modification is a relatively simple electrical circuit change and would not affect the charger package.

The non-recurring cost of modifying the ELMS charger minimum input voltage to improve efficiency and to incorporate charge control for three batteries is approximately $100K. The reduction in solar array cost and the improved flexibility would justify making these modifications.

1.3.3.2.2.2 Battery Alternatives

The basic energy storage configuration and requirements that were used to evaluate candidate batteries were;

- Twenty-two series connected NiCd cells per battery
- Use two or three batteries to simplify battery charger and other power module changes
- Provide total capacity option of 40 to 120 A.H.
- Use existing designs, if possible.

All of the above energy storage requirements can be satisfied with existing or slightly modified, 20, 36 or 45 A.H. batteries. Table 1.3.3-5 identifies a number of existing or slightly modified batteries. In the case of the OAO battery, a packaging option exists where the present two assembly - 3 battery configuration is reduced to a single, 22 cell package by shortening of the battery rails.

A comparison of the recurring cost per Ampere Hour and weight of the candidate batteries indicates that weight and cost savings can be realized by using other than OAO batteries. By comparison, a modified version of the Grumman ELMS battery, which is a 22 cell 20 A.H. battery that utilizes cells identical to those in the OAO battery, would cost $20.6K per 20 A.H. or $61.8K per 60 A.H. as compared to an OAO battery cost of $85K per 60 A.H. In addition to the extra cost of the OAO battery, the OAO battery set weight of 164 pounds is approximately 5 pounds heavier than 3-Grumman ELMS batteries.

A relatively new battery that has been developed by Eagle Picher for the Philco Ford NATO III spacecraft promises even better cost and weight savings. Three, 22 cell, 20 A.H. batteries using the new lightweight battery cells could result in about 48 pounds of weight saving as compared to the OAO battery. The recurring cost per 20 A.H. battery is only $18.2K, however, the non-recurring cost to qualify the battery for EOS and obtain life data would be approximately $100K.

Table 1.3.3-5 EOS 22 Cell, Ni-Cd Battery Alternatives

Battery	Mfg.	Rated Capacity AH	Weight Per Cell (lbs)	Battery Weight (lbs)	Weight Per A-H (lb/AH)	Dimensions (Inches) L W H	Recurring Cost	$/AH
OAO, 3*20 AH	Gulton	3*20 AH	2.1	164	2.73	2*(18 5/8 x 11 x 10)	85K	1.42K
Modified OAO-single 20AH	Gulton	20 AH	2.1	54.7	2.73	12.6 x 11 x 10	30K	1.50K
Modified ELMS	Gulton	20 AH	2.1	53.1	2.66	12.6 x 7.4 x 7.4	20.6K	1.03K
Modified SAR 8022-5	Eagle Picher	20 AH	2.3	65.8	3.29	20.5 x 8.5 x 6.5	13K	0.65K
Modified NATO III	E-P	20 AH	1.28	32.4	1.62	12.4 x 7.5 x 5.3	18.2K	0.91K
Modified SAR 8055-19	E-P	36 AH	2.80	89.9	2.50	19.8 x 8.5 x 7.0	17K	0.47K
SAR 8054	E-P	45 AH	3.60	103	2.29	22.3 x 9.0 x 8.2	18K	0.4K

When stored energy requirements are in the 60 to 120 A.H. range, either a 36 A.H. or 45 A.H. battery would be used. On the basis of $/AH these size batteries provide the most economical approach to satisfying large capacity requirements.

1.3.3.2.2.3 Misc. Power Module Alternative

The components and functions required in the Power Module other than the energy storage and control and solar array control functions, are all basically dictated by the general configuration of the subsystem and interface, command, telemetry and monitoring requirements. Such requirements as current sensing, signal conditioning, bus protection do not involve any major design-cost trades but rather only an optimization of components, location and packaging.

Of special interest were the approaches to implementing the requirements for module/spacecraft electrical connectors suitable for on-orbit resupply.

The major goals considered for selection of a resupply disconnect are as follows:

- Existing hardware (no development)
- Proven performance
- Low cost, weight
- Rugged construction
- Environmental

See Table 1.3.3-6 for connector type comparisons.

Table 1.3.3-6 Resupply Connector Candidates

CONNECTOR TYPE	ADVANTAGES	DISADVANTAGES	VEHICLE EXPERIENCE
F-14A Weapons Rail Electrical Umbilical	Rugged, proven design	High mating forces req'd; high cost (relative)	F-14A Weapon Rail
Zero Insertion Force	No mating force	No hardware; high contact resistance	None - Requires development
Rack/Panel Types - Cannon DPK, DPD	Low cost (relative)	High mating force; require mechanical guide pins; DPD type non-environmental	DPK - APOLLO, Various aircraft

1.3.3.2.3 Solar Array Alternatives

One of the major considerations in selecting the overall EPS configuration was the flexibility in the electrical interface between the power module and the solar array. This interface flexibility provides latitude in defining a solar array that is optimized to particular mission requirements.

Major trade-offs that are generally considered when defining a spacecraft solar array include:

- fixed vs oriented
- rigid vs flexible

In general fixed arrays are preferred over rotating array from the standpoint of reliability and vehicle dynamics. However for some missions, solar incidence angles could result in the need for a very large array area and additional cost and weight compared to an oriented array.

In the case of EOS, the basic candidate orbits are sun-synchronous at altitudes between 300-500 nautical miles with descending node times of day (DNTD) ranging from 9:30 AM to 12 noon. Furthermore, the selection of the DNTD should be left as a variable, with final selection occuring as late as when the spacecraft is "on the pad" ready to go.

The question of fixed array versus an oriented array for EOS is most easily answered by considering the noon-midnight orbit requirement. For this condition the sun is nominally in the orbit plane and all configurations of fixed arrays would require solar array areas of at least 2½ to 3 times the area required by a single axis rotating array. Assuming an array cost of \$3K/ft^2 and an orientation drive cost of \$150K the rotating array would be the more cost effective approach for solar array areas in excess of approximately 50 ft^2. Since all prime EOS missions will require in excess of 100 ft^2, the oriented array is the only logical choice, if cost is the prime consideration. However, rarely is the cost the sole criteria in selecting a spacecraft solar array. Weight, deployment/retraction, stowage volumes, vehicle dynamics, and other overall spacecraft system considerations must be incorporated into the selection of the solar array.

In the relatively recent past, significant attention has been directed toward reducing the weight of solar arrays. The conventional approach to deployed arrays was the use of a rigid substrate material such as honeycomb. The rigid substrate array had the advantage of minimizing spacecraft dynamic interaction with the S/C control system and could be produced at a recurring cost in the order of \$3K/ft^2. The main disadvantage of this type array is that weight is rarely less than 1 pound/ft^2.

The approaches to significantly reducing the array weight have generally all employed a flexible substrate material that is stowed in a flat-pack (Z) arrangement or a drum-roller similar to a window-shade. Weight reductions to less than 1/3 the weight of the rigid array have been realized however the recurring cost/ft^2 have generally at least doubled.

Table 1.3.3-7 describes a number of existing rigid and flexible arrays which are typical of those that could be considered for EOS. Considering the wide range of choices available, it would appear likely that the use of an existing solar panel or array design could be used for EOS with potential cost benefits.

Table 1.3.3-7 TYPICAL SOLAR ARRAY CANDIDATES SUITABLE FOR EOS

Solar Array Type/Description	Size	Weight	Power	Recurring Cost	Remarks
Req'd Flat Panel - Xerox P=95 0.300 alum honeycomb with alum face sheets	17 x 66 in 7.8 ft^2	7.8 lbs (1.0 lb/ft^2)	92.5 watts @28°C (11.9 w/ft^2)	\$23K \$2.95 K/ft^2	
Req'd Flat Panel - Xerox Timation IIIA (alum honeycomb)	54" x 21.75" (8.2 ft^2)	17.2 lbs (2.1 lbs/ft^2)	96 W @ 28°C (12 W/ft^2)	37K \$4.5K/ft^2	
Solar Cell Modules - Spectrolab ATM (alum honeycomb with alum face sheets)	20" x 24.63" (3.4 ft^2)	est. 5 lbs ~1.5 lb/ft^2	35 W @28°C (10 W/ft^2)	approx \$10K \$3K/ft^2	natural freq 0.25 Hz
Flexible Roll-Up Solar Array-Hughes FRUSA laminated Kapton-H film and fiberglass substrates (Flown on USAF mission, 1971)	DRUM 5 1/2 ft x 8" dia array - two blankets each 5 1/2 ft x 16 ft (166 ft^2)	70 lbs Druam Assy 36 lbs 0.205 lb/ft^2 + drum	1600 W (~10 W/ft^2)	\$1.2M	
Flexible Roll-Up Solar Array - GE (RA250)	Drum 8.2" x 8" Dia array 8.2" x 34 ft	82 lbs Drum 36 lbs Blanket 46 lbs (.19 lbs/ft^2 + drum)	(2500 watts) 10 W/Ft^2)	Not Available	natural freq. less than 0.1 Hz

Solar Array Drives - Alternatives

For EOS missions, where solar array orientation is advantageous, a number of existing systems could be utilized with modification.

Table 1.3.3-8 describes some of the Solar Array Drive requirements with respect to mission pecularity/cost impact.

TABLE 1.3.3-8 Solar Array Drive Requirements

PARAMETER	REQUIREMENT	MISSION PECULIAR MIN-MAX	MISSION PECULIAR COST IMPACT
Operating Voltage	28 ±7VDC	-	-
Operation	Continuous Bi-Directional	-	-
Modes	Track, CW/CCW Fast Slew CW/CCW	-	-
Trace Speed	-	3.6 Degrees/Min Nominal	None
Fast Slew	-	10-15 Degrees/Min.	None
Position Indicator	± 1 Degree	-	-
Torque (Inertia Load)	-	TBD	Minimal
Power Transmission	-	15-75A, 30-50VDC	Minimal

During sunlight periods of the orbit, the solar array will be driven to maintain optimum solar incidence. During the eclipse or dark periods, there exists solar array drive alternatives:

A - Continuous rotation of the array (single direction)

B - Limited rotation of the array with automatic reverse fast slew (start and stop) into proper orientation for entering back into light.

The optimum rotation selection for the EOS is a continuous drive compatible with the sensitivity of the attitude control subsystem. The major determinant for this tradeoff is the necessity to minimize resultant disturbance torques created by periodic stops, starts, and reversals. Because of this factor, the drive will be driven via the On Board Computer with updates as required to reduce angle errors in lieu of the standard sun sensor closed loop system. See Para. 1.3.1.4 for specific requirements.

Various motor types were evaluated. The major influencing factors for determination and selection are: reliability; lowest risk (life limited components); low cost and weight; successful space application; and growth potential (redundancy, increased torque requirements). See Table 1.3.3-9 for motor type comparisons.

The alternatives of power and signal transmission across rotating joints included slip rings, flexcable, power clutch, rolling contacts, and rotary transformers. The last three methods were eliminated as high risk items because development is required and hardware, test data, and flight experience are lacking.

Flexcables are considered as first choice ahead of slip rings but are restricted to limited rotation drive systems. Since the influencing trade-off primarily concerns limited versus continuous rotation and not specifically transmission methods, it can be concluded that slip rings will be utilized to transmit all electrical interfaces between the solar array and spacecraft. Slip ring brush, ring, and lubricant materials are considered state of the art and have accumulated many successful hours in space applications. See Table 1.3.3-10 for slip ring and flexcable comparisons.

Based upon the foregoing, several existing drive systems were compared and reviewed for EOS requirements. The prime considerations for candidate selection are low cost, adaptability for modifications to meet all parameters and requirements and the adaptability toward growth potential (redundancy and resupply). Table 1.3.3-11 presents a comparison of candidate drive systems.

Table 1.3.3-9 - Motor Type Tradeoffs

MOTOR TYPE	WEIGHT (LBS.)	SIZE	TORQUE EFFICIENCY FT-LBS/WATT	CONTROL SYSTEM COMPATABILITY	LIFE (Does Not Include Bearings)	REMARKS
DC Brush Type Torque	7.5	8" dia x 1-3/4"	0.027	Excellent	Brushes are only wear elements. However, 10 yrs is feasible	Can handle large inertia loads. Impressive history of successful space applications
DC Brushless Type Torque	4	4-3/4" dia x 2-1/4	0.022	Electronic Complexity Otherwise Excellent	10 yrs feasible	Can handle large inertia loads. Very complex electronics
Servo Motor	2=motor 5=gearhead	No unit mfgr with torque rqmts except with extra gearhead	0.00025 at motor	More complex electronics to control AC	10 yrs feasible	Poor torque per watt ratio. Inverter losses must be charged against drive
DC Stepper	4	4-3/4" dia x 2-1/4	0.022	Complicated electronics	10 yrs questionable due to repeated mech shock	High surge currents, does not drive inertia loads well. Detent torque could be useful
AC stepper	3-motor 7-gearhead	No unit mfgr with torque rqmts except with gearhead	0.00002	Complicated electronics	10 yrs questionable due to repeated mech shock	Poor torque efficiency, high surge currents does not drive inertia loads well

Table 1.3.3-9 - Motor Type Tradeoffs (Cont.)

MOTOR TYPE	WEIGHT (LBS.)	SIZE	TORQUE EFFICIENCY FT-LBS/WATT	CONTROL SYSTEM COMPATABILITY	LIFE (Dord Noy Include Bearings)	REMARKS
Induction	20	10"dia 11" long	0.0007	More complex Electronics to control AC	10 yrs feasible	Low starting torque, high starting surge current. Inverter inefficiency must be charged to drive
AC Synchronous	25	10" dia 14" long	Poor - depends on starting method	More complex electronics to control AC	10 yrs feasible	Poor starting torque without aux. means. Inverter losses must be charged to drive.
AC Torque	40	15" dia x 4	0.002	More complex electronics to control AC	10 yrs feasible	Poor torque per watt ratio. Poor torque per pound ratio. Inverter losses charged to drive.

TABLE 1.3.3-10 COMPARISON OF SLIP RING AND FLEX CABLE SYSTEMS (EOS)

		Slip Rings	Flex Cable
	Application	360° continuous rotation	Limited rotation
	Space Experience	Many - only continuous operation	Many - limited rotation only
	Restriction	Limited rotation creates pile up of contact brush material	Requires positive stop to prevent damage to wire
D	Lubricant	Selection important for space application	Not applicable
E	Efficiency	Contact resistance (friction) reduces efficiency	Reduced only by copper feeder resistance
S	Limited Life Components	Sliding electrical contact wear	None
I	EMI	Sliding contacts create noise	None
G	Torque Requirements	Torque to overcome contacts is constant	Torque to overcome flex leads not constant
N	Radiation Hardening	Contact material/finish, dielectric, lubricant, & wire insulation selection are important	Wire insulation selection important
	Life Testing	Oscillation rate important because of critical combination of contact/brush material/finish and lubricant	Can be accelerated for quick quantitative results
	Weight	$\geq$ 1 lb heavier than flex system	Design simplicity reflects lighter weight & lower costs.
	Cost	Higher than flex system	
	Reliability	Failure rate $860/10^{9}$ hr at 25 percent rated stress and 30°C per brush/slip-ring contact	No published numbers - but system contains no sliding contacts & has no material/finish selection criticality

Table 1.3.3-11 Solar Array Drive Candidate Comparison

SELLER	BALL BROS	BENDIX	HUGHES	TRW
Model (Program)	OSO Components	ASAD (Nimbus) ELMS	FRUSA	NIMBUS/ERTS
Existing/Mod/New	New	Mod	Mod	Mod
Non Recurring $	85,698	88,993	891,615	Non avail.
Recurring $ Each S/C	107,838	51,940	213,854	280,800
Cost Risk	Low	Low	Med	High
Adaptability to EOS Requirements	Good	Excellent	Fair	Fair

1.3.3.3 Selected Configuration

The basic EPS system configuration selected for further study and definition is shown in Figure 1.3.3-2. This preferred EOS-EPS configuration is functionally the same as the GSFC-OAO power system configuration and was selected for EOS for the following reasons:

- All major power functions can be satisfied with existing or modified components and therefore reducing or eliminating development risks.
- The system offers the most flexibility to be optimized to specific mission requirements without major changes to power module components, harness or interfaces.
- The system satisfies all EOS requirements for a self-contained, standardized, modular power system with multi-mission capability.

The specific components of the Power Module are itemized in Table 1.3.3-12 along with weights and cost.

The selected series regulator/battery charger is the ELMS charger modified to lower the minimum input voltage allowed and the addition of circuitry for independent charge

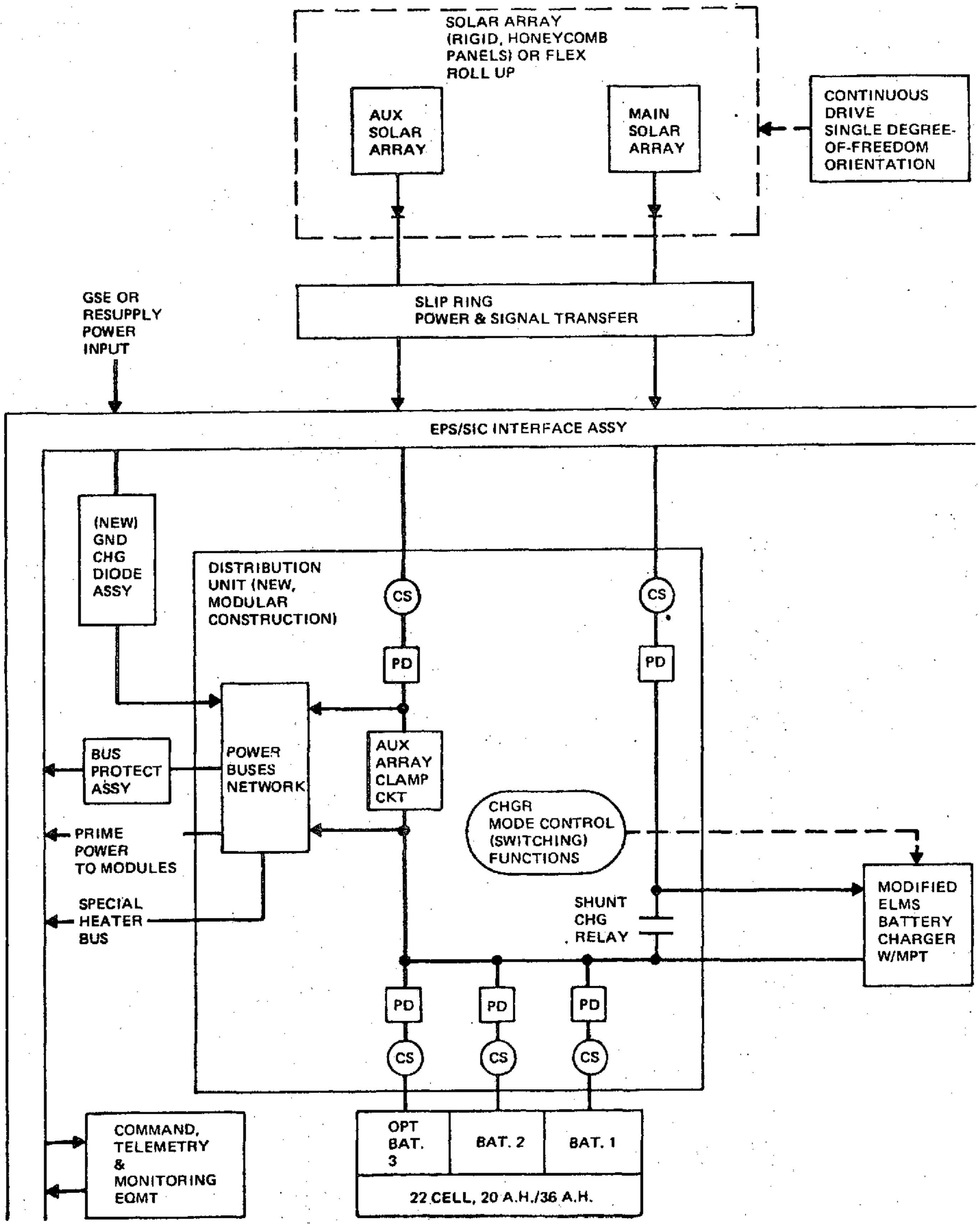

Fig. 1.3.3-2 Selected EPS Configuration

control of up to 3 batteries. This charger, which is manufactured by Gulton-Engineered Magnetics Div, is depicted in Figure 1.3.3-3 and was selected on the basis of the following:

- Lower weight and cost compared to the OAO equipment
- Modular construction facilitating repair, maintenance, and optimization to mission peculiarities
- Use of 5 basic power regulator modules which are operated in parallel. The number of power regulator modules can be reduced to maintain optimum efficiency when spacecraft power and redundancy requirements allow
- Current limiting protection of power regulator modules
- Self-contained, solar array maximum power tracker with dual-redundant electronics
- Flexibility in allowable input voltage range from 35 to 125 V.D.C.

TABLE 1.3.3-12 SELECTED EPS COMPONENTS

Power Module	Recurring Cost, $K	Weight, lb	
Battery 2-22 Cell, 20 AH	37	64	Modified NATO III
Battery Charger	55	24	Modified ELMS
Central Power Cont. Unit	50	23	New - Grumman
Signal Cond. Assy	25	10	
Grd Chg Diode Assy (1)	5	7	New - Grumman
Bus Protect. Assy	10	5	New - Grumman
S/C Interface Assy	10	12	New - Grumman
Bus Assy (3)		2	New - Grumman
Connectors	4	4	New - Grumman
Wiring & Misc		18	New - Grumman
Remote Decoder			
Dual Remote MUX	10	2	New
Power Module Total	206	171*	

*Excludes weight of:
- Power Module Structural frame
- Battery heat sinks, and louvers

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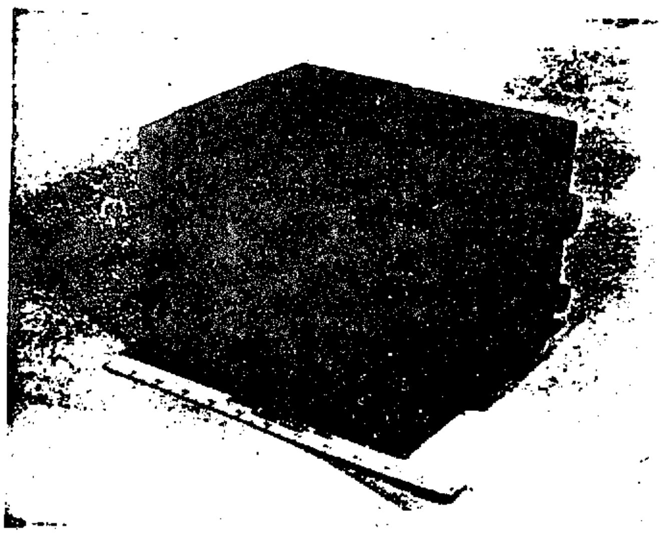
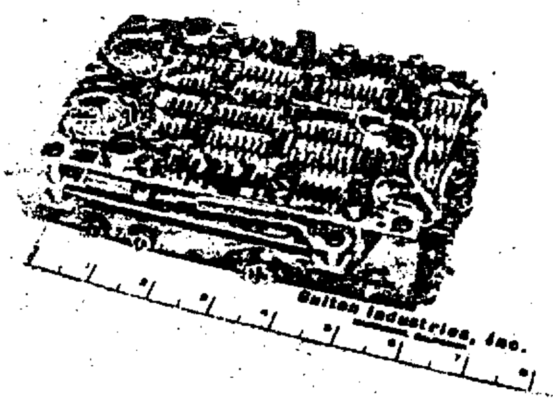
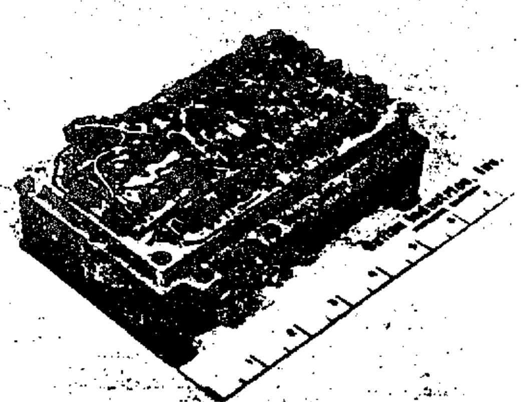


PEAK POWER TRACKER AND BATTERY CHARGER



CONTROL MODULE



POWER REGULATOR MODULE

FIGURE 1.3.3-3
Selected Peak Power Tracker and Battery Charges (Modified ELMS unit manufactured by Gulton-EM.)

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The battery capacity option will be satisfied by two different capacity batteries; a 20 AH and 36 AH size configured as single 22 cell batteries. The use of two capacity batteries will minimize changes to the basic power module components and result in a more predictable load sharing between batteries.

For the 20 AH battery, the modified NATO III battery is selected strictly on the basis of low weight and cost. However, since this battery is relatively new with a limited amount of life data, a battery consisting of Gulton OAO type cells will be retained as an alternative.

For the larger size batteries, either the modified 36 AH or 45 AH battery would be cost effective; however, preference is given to the 36 AH size since the cell has flight history and is less risky in its thermal design.

In order to implement the requirements for power disconnect of all power sources, current sensing provisions, battery and battery charger control functions and power distribution networks, a new Central Power Control Unit or distribution unit is included. This unit provides a central point for accommodating mission peculiar solar array, battery and load interfaces, and will feature modular design and construction.

The requirement for modular resupply requires special connectors for the spacecraft/module interfaces.

The F-14A weapons rail umbilical connectors which is depicted in Figure 1.3.3-4 meet or exceed all the key parameters and goals for EOS resupply requirements and will be used as a standard disconnect for all modules in a resupply configuration:

- Currently being qualified for F14
- Cost approximately $1000 per mating pair
- Weight of mating pairs varies between 1.1 and 2.23 lbs depending on model/insert arrangement
- blind mate capability (0.150 inch maximum misalignment)
- Anti-blind roll off shells
- 1 inch axial movement of structure without affecting continuity, environmental seals or RFI continuity

GRUMMAN

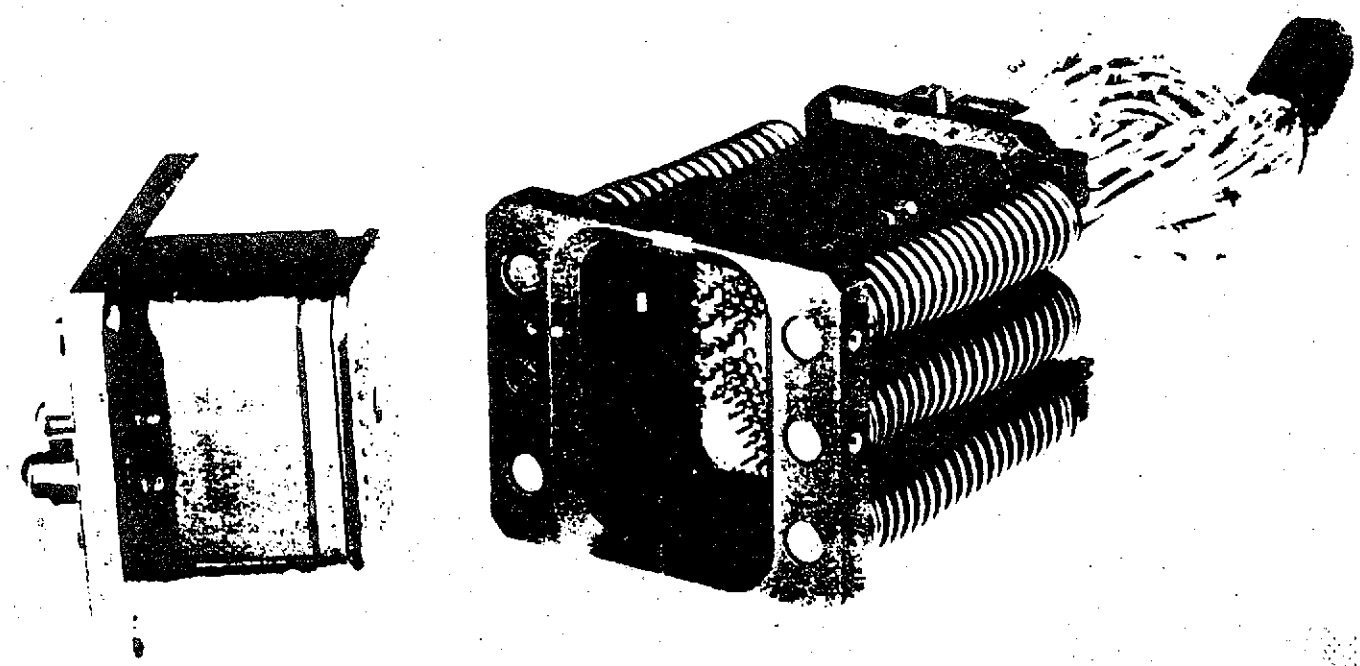
TRADE STUDY REPORT

TITLE	TRADE STUDY REPORT NO. 1.3.3
	WBS NUMBER 1.7.3.2/1.7.3.7

FIGURE 1.3.3-4

BLIND MATE UMBILICAL

F-14-A WEAPONS RAIL



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*GAC 3711 3-72

- o Connector meets NAS 1599 and NAS 1600 requirements
- o Insert arrangements optional - includes coaxial contacts, over 100 contacts available per insert
- o 360-degree RFI continuity
- o No lock disconnect mode - axial tension

The preferred solar array configuration for the basic EOS missions is where the entire array is co-planer and mounted on one side of the spacecraft and is continuously rotated about the spacecraft pitch axis. To allow for variation in the descending node time of day (DNTD), a tip adjustment is provided which allows changing the angle formed by the rotating shaft and the plane of the array. For the EOS DNTD range of 9:30 AM to 12 noon, this angle will range from 0^{o} at 12 noon to approximately 45^{o}.

System aspects of selecting either a rigid or flexible, roll-up type array have not conclusively established a preferred approach, therefore both types of array will be retained as alternatives for EOS.

Since a multitude of EOS missions were being considered in parallel with the EPS efforts, it was necessary to evaluate the solar array sizing on a general rather than detailed basis.

Solar array size estimates were based upon the following:

- o Power Module similar to GSFC demonstration model EPS, except PCU, PRU and Diode Box were replaced by the ELMS battery charger
- o An auxiliary (DET) array was not used
- o Peak loads exceeding the solar array power capability were not considered
- o The solar array was rotating and oriented within 5^{o} of normal incidence during sunlight periods
- o Solar array power degradation was 12% which corresponds to the predicted degradation after 2 years in a 500 nm, 90^{o} incline orbit. (nominal 14 mil thick, 2 ohm-cm N-on -P silicon cell with 12 mil fused silica cover slides)
- o Light/Dark ratic corresponding to a worst case 300nm/noon-midnight orbit was used.

Based upon the above, the solar array area required will be approximately $\frac{1}{4}$ ft^2 for each orbital average watt of load. Variations in detailed design parameters and optimizing the system configuration/operation for a particular mission (e.g. using an auxiliary array) could result in a tolerance of this $\frac{1}{4}$ ft^2/watt value of 10 to 15%.

Therefore for the EOS orbital average power requirements of 500 to 1000 watts, solar array areas ranging from 125 to 250 ft^2 will be required.

The prime candidate for solar array drive is the Bendix unit. This unit meets all design considerations including low cost, continuous rotation, and adaptability to other required modifications. The motor is a brushless DC torquer which exhibits high reliability, low risk (no life limited components), lightweight, and proven space experience. Particularly impressive with the candidate drive is the adaptability of the Bendix motor and assembly to redundancy techniques for optimum reliability. In addition, available torque is more than adequate to handle physically worst case array configurations.

Growth potential of the selected drive was considered with respect to redundancy techniques and resupply.

Redundancy Techniques - The selected drive is adaptable for improved reliability by utilizing several redundancy techniques. See Table 1.3.3-13 for redundancy impact.

TABLE 1.3.3-13 Redundancy Techniques

REDUNDANCY	DEVELOPMENT	RELATIVE IMPACT COST	WEIGHT
Electronics	State-of-the-Art	Minimal	~2 lbs
Dual Stator Windings	State-of-the-Art	Minimal	<1 lb
Bearing Within Bearing	MMA life testing will will prove design-bearings are State-of-the-art	Minimal	<1 lb

Resupply - Since the selected drive system is classified as an MMA, the consideration for replacement during a resupply mission configuration is logical. In this configuration, the drive assembly is attached to a structural interface unit having electrical disconnects. Additional weight is estimated at approx. 6.0 lbs and hardware costs <$1000.

D 1.3.4 Propulsion

D 1.3.4.1 Orbit Adjust Subsystem (OAS)

D 1.3.4.1.1 Requirements Definition

A number of requirements have a significant influence on the design of the orbit adjust subsystem (OAS). Size, configuration, function and cost are impacted by the following:

- o Launch vehicle orbit injection errors
- o Operational altitude
- o Modularity

The major influences on sizing the OAS are the launch vehicle orbit injection errors. Table D 1.3.4-1 lists the velocity and position errors which result from the use of Titan and Delta launch vehicles. It should be noted that the listed velocity errors are the RSS values of the injection velocity and inclination errors. The propellant required to correct these injection errors represents approximately 97% of the total translational propellant on-board the S/C.

INJECTION ERROR	TITAN	DELTA
Velocity, ft/sec	42	28.6
Position, nm	1.5	15

Table D 1.3.4-1 Launch Vehicle Injection Errors

The spacecrafts operational altitude affects the OAS design in two ways. The first is the propellant required to perform the orbit keep function varies with altitude, increasing at the lower altitudes. For the orbit selected for our S/C the orbit keep propellant represents 3% of the translational propellant

The second affect only becomes a consideration when the operational altitude is in excess of $\simeq$ 400 nm. When the orbit exceeds $\simeq$ 400 nm it becomes necessary to incorporate kick stages to achieve the desired orbit. This imposes requirements on the OAS to perform a thrust vector control (TVC) function during the SRM burns and to correct injection errors caused by SRM variability. Since the orbit selected as a result of our studies is 366 nm, this latter requirement drops from consideration for this configuration.

The requirement for modularity and patentially resupply results in the OAS being installed in a separate structure mounted to the aft end of the S/C. From a propulsion point of view, this is the ideal location in that the thrusters are mounted to provide the desired thrusting along the vehicle flight path. The modular approach also eliminates the need for fluid interfaces between the main S/C structure and the thruster pods. Finally, the modular arrangement, mounted on the aft end of the S/C, assures that the maximum distance possible is attained between the thrusters and the solar array and instruments., thus, minimizing the possibility of impingement or interaction with the thruster exhaust plume.

In general rocket plumes from such sources as OAS thruster exhaust, etc., can create some undesireable effects on vehicle/subsystems operations. However, because of the aft locations of the OAS units these effects are likely to be non-existent for the suggested configurations. As an example consider the effect of the NH_3 exhaust products from hydrazine burning as a possible IR absorber. Ammonia is known to have a strong absorption band in the region of the TM band 7 and could conceivably affect the quality of the IR imagery in this band if the gas density in the line of sight is great enough. A quick look at this problem has indicated that the exhaust gases should leave the vicinity of the vehicle in 1 to 2 seconds and therefore present no problem of this kind.

D 1.3.4.1.2 Configuration Alternates

Two alternatives, one using hydrazine ($N_2 H_4$) and the other gaseous nitrogen (GN_2), were considered to fulfill the OAS function. The results of the trade study are shown in Table D 1.3.4-2

	$N_2 H_4$ System	GN_2 System	Delta ($N_2 H_4$ Vs. GN_2)
Weight, Lb.	30.2	170.3	- 140.1
Cost, $K	148.0	121.0	+ 27

Table D 1.3.4-2 $N_2 H_4$ Vs. GN_2 Orbit Adjust Subsystem

While the GN_2 fueled system provides a less complex and lower cost OAS, it is a much heavier system. Since weight is a major consideration in the Delta 2910 S/C configuration, the lighter weight $N_2 H_4$ system was selected. It should be noted that the weights shown above include only component and propellant weights. The structural weight penaltyassociated with the GN_2 system is not included.

D 1.3.4.1.3 Selected Configuration

The selected orbit adjust subsystem is a hydrazine fueled system utilizing four 5 lb thrusters and operating in a blow-down mode. The equipment is mounted in a module mounted on the aft end of the S/C, see Figure D1.1.1-15 .

A single thread, non-redundant subsystem, shown schematically in Figure D 1.3.4.1 has been selected for the orbit adjust subsystem. The $N_2 H_4$ tank design is a 9.4 in diameter, 420 cu. in. tank previously used on the AEROS program. Positive expulsion of the 10 lb of $N_2 H_4$ in each tank is provided by an AFE-332 diaphragm. The system operates in a blowdown mode starting at ≈ 400 psi and ending at 130 psi. The filter and latching solenoid valve designs have been used on Intelsat IV; the fill disconnect designs are from the Viking program. The 5 lb. thruster design has flown on ATS-C, IDCPS/A and NATOSAT satellites and has been qualified for Skynet II, CTS and the NRL Multiple Satellite Dispenser (MSD). The thrust level is nominally 5 lb. However, with the tank pressure profile described above, the range of thrust level is 5.5 lb. down to 2.6 lb, see Figure D 1.3.4-2.

Adding redundancy to the subsystem to provide fail-safe operation requires the addition of two latching solenoid valves and a second solenoid/seat assembly to each of the thurster valves, see Figure D 1.3.4-3 A total weight increase of 3.22 lb. with a corresponding increase in cost of $11K provides fail-safe redundancy

The selected orbit adjust subsystem provides the flexibility to accomodate vehicle growth or mission changes. The two tanks used are capable of storing an additional 3 lb.

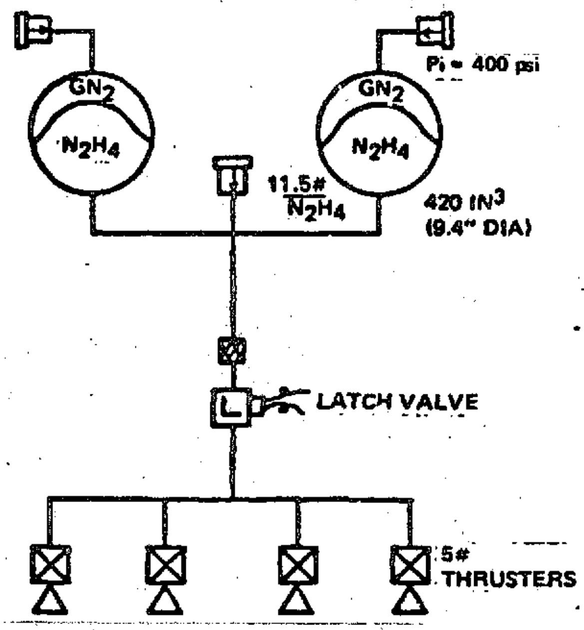


Fig. D.1.3.4-1 Hydrazine Orbit Adjust Subsystem (Single Thread)

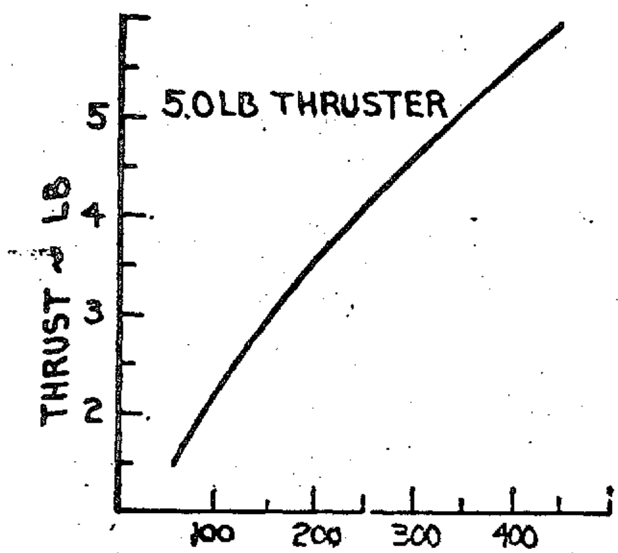


Fig. D.1.3.4-2 Tank Press PSIA Thrust vs Tank Press

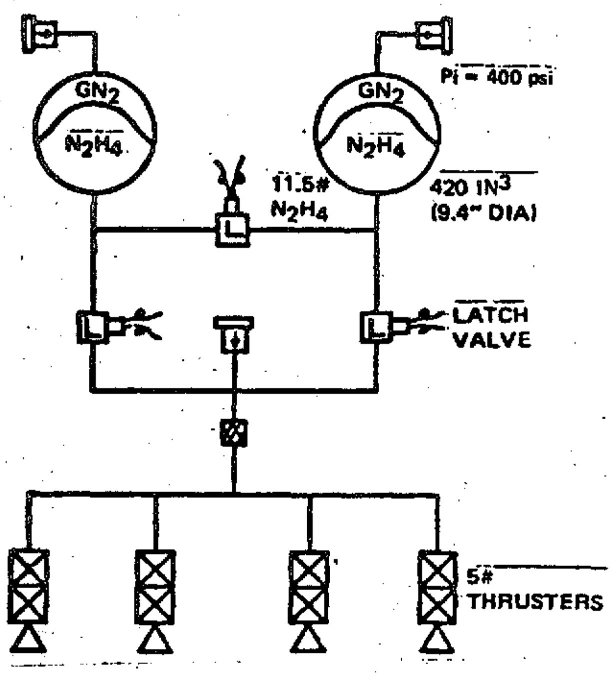


Fig. D.1.3.4-3 Redundant Orbit Adjust Subsystem of N_2H_4. The structural arrangement of the OAS module is such that is is also possible to mount and connect two additional tanks into the system. Thus, a total increase in propellant weight of 26 lb. can be provided. It is well within the capability of the thruster design to burn this additional propellant. Assuming that on the average 2 of the 4 thrusters are firing, the thruster firing time is increased by slightly less than 600 seconds.

D 1.3.4.2 Reaction Control Subsystem (RCS)

D 1.3.4.2.1 Requirements Definition

The requirements having significant influence on the design of the reaction control subsystem (RCS) are:

- o Vehicle stabilization and restabilization
- o Wheel desaturation
- o Spacecraft modular/structural arrangement.

Each of these impact the size, configuration and cost of the RCS.

As can be seen from Table D 1.3.4-3 vehicle stabilization and restabilization have a small impact on the total propellant loading. However, the need for vehicle stabilization initially and during injection error firings establishes the 1.0 lb thrust level. The burn times using smaller thrusters (e.g., 0.1 lb thrusters) would be excessive.

The greatest impact on propellant loading results from the need to provide some of the reaction wheel desaturation with reaction jets. Wheel desaturation requires $\approx$73% of the rotational propellant. The quantity of wheel desaturation propellant is based on performing 20% of the total desaturation using reaction jets.

The requirement for very low impulse bits for desaturation established the need for low thrust level thrusters on the order of 0.05 to 0.1 lb of thrust. Analysis showed that the minimum impulse bit (MIB) capability of existing 0.1 lb thrusters (0.002 lb-sec) is acceptable for wheel desaturation. The 0.1 lb thruster was, therefore, selected.

Table D.1.3.4-3 Mission Impulse Requirements

MISSION PHASE	Impulse (Lb-Sec)					
	Translation			Rotation		
	X	Y	Z	R	P	Y
Initial Stabilization				7	14	14
Correct Injection Error	4070					
Control During Correction For Injection Error				0.5	21	21
Stabilize After Solar Array Deployment				1	2	2
Orbit Keep	140					
Gravity-Gradient, Jets				105	236	0
Totals	4210			113.5	273	37
Contingency 10%	421			11	27	4
Overall Totals	4631			125	300	41
Total 5097						

As is the case with the OAS, the requirement for modularity results in the RCS being installed in a separate structure mounted to the aft end of the S/C. The module provides the means of easily orienting the thrusters to provide pitch, yaw and roll control. However, because the module is on the aft end of the vehicle, pitch and yaw firings result in small translational movements of the vehicle in addition to the desired rotation. The question of the acceptability of these translations will be addressed in the "Coupled vs Uncoupled Pneumatics" (RCS) study to be performed during the latter part of our study.

Matching the propulsion module, which desires to have the thrusters acting along orthogonal axes, to the triangular shaped main S/C structure presents problems in providing sufficient moment arms for the thrusters. The result is a module structure incorporating outriggers as shown in Figure D1.1.1-13. As can also be seen, the thruster pods are set at 45° to the vehicle Y-Z axes. However, the mounting of the thrusters within the pods is such that the desired orthogonal arrangement is maintained.

D 1.3.4.2.2 Configuration Alternates

Two alternatives were considered to fulfill the reaction control (pneumatics) subsystem (RCS) function. The first of these shown schematically in Figure D 1.3.4-4 assumed the use of GN_2 as the propellant. The GN_2 system is a simple design carrying 11.8 lb. of GN_2 with the capability to provide initial stabilization and restabilization of the vehicle as well as its allotted wheel desaturation requirement. The logical alternative to using GN_2 was the use of hydrazine as the propellant. Since the vehicle is already carrying a hydrazine fueled OAS, it follows that combining the reaction control subsystem with the OAS should be considered. The all N_2H_4 reaction control/orbit adjust subsystem is shown in Figure D.1.3.4-5.

In order to make an equal comparison, the combined GN_2 reaction control and N_2H_4 subsystem weights and costs were compared to the all N_2H_4 subsystem. The results of the trade study are shown in Table D 1.3.4-4.

	Combined GN_2/N_2H_4	All N_2H_4	Delta (Combined Vs. All N_2H_4)
Weight, lb.	78.2	40.2	- 38
Cost, $K	310.4	336.4	+ 26

Table D 1.3.4-4 GN_2 Vs N_2H_4 Reaction Control Subsystem

On an individual basis it appears that the GN_2 reaction control system is lower in complexity as well as in cost. However, when the total propulsion module is considered the N_2H_4 reaction

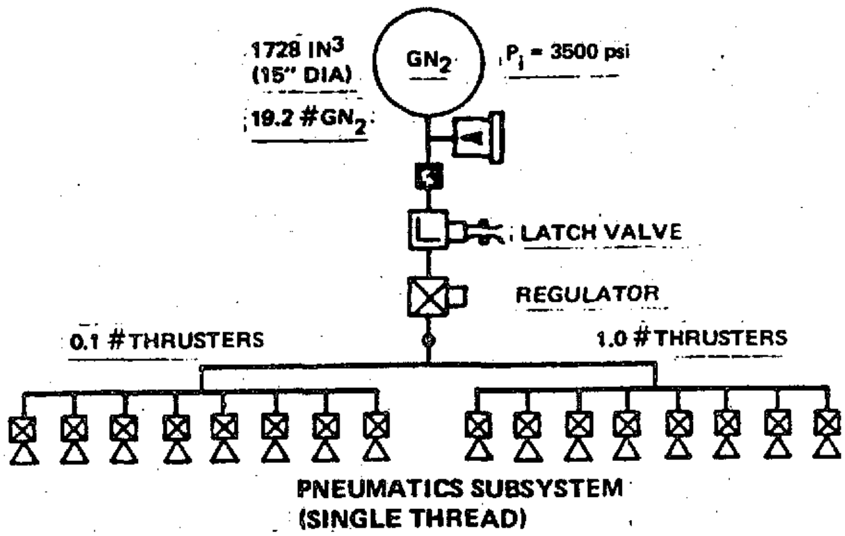


FIGURE D 1.3.4-4

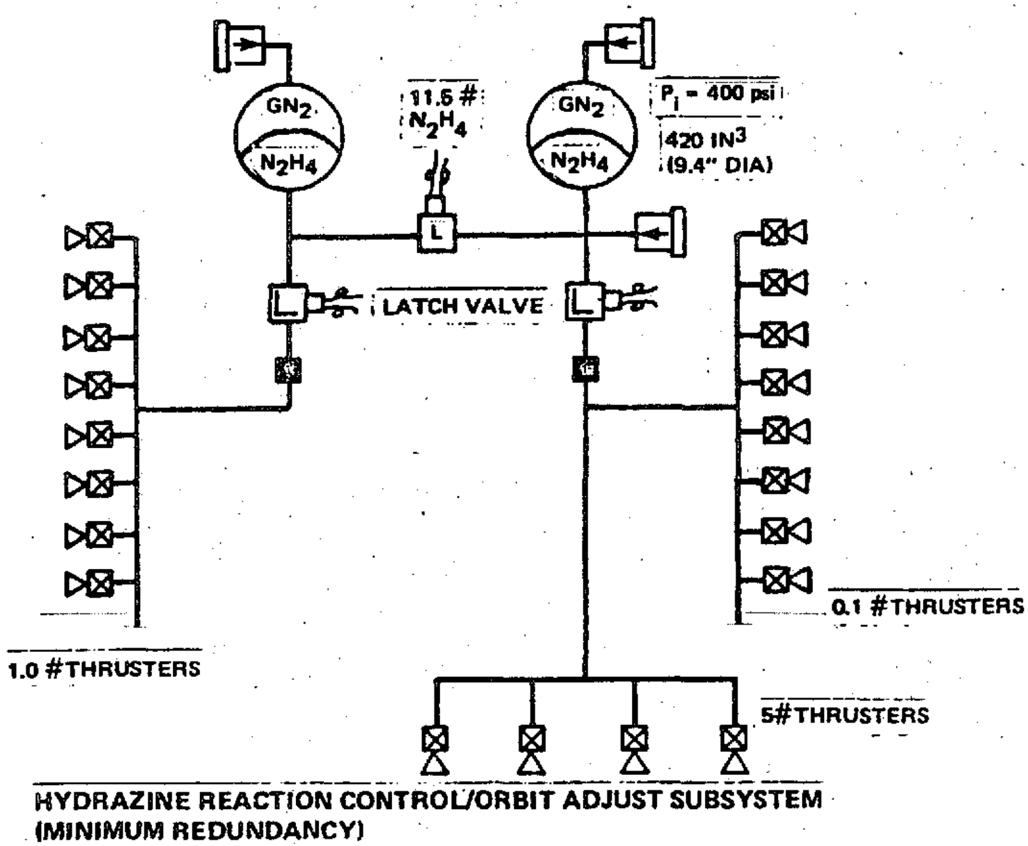


FIGURE D 1.3.4-5

control/orbit adjust subsystem is the least complex system. The number of tanks is reduced because the OAS tanks are off-loaded and have sufficient volume to absorb the RCS propellant ($\approx$3 lb), thus, the GN_2 tank is eliminated. The GN_2 regulator is also eliminated. Another consideration is a reduction in safety hazard. The 3500 psi GN_2 operating pressure is eliminated. The N_2H_4 system operates at $\lesssim$400 psi. The final consideration in the selection of the RCS is the reduction in total program costs which results from the development of a single RCS/OAS design. The reduction in design engineering manhours alone reduces the delta-cost between the two systems by $40K. Further savings in procurement, testing, assembly and check-out costs estimated to be $46K are realized from the use of a single design. Including these latter cosiderations and their associated cost reductions in the overall assessment of the reaction control subsystem led to the selection of the N_2H_4 fueled RCS/OAS.

D 1.3.4.2.3 Selected Configuration

The selected reaction control subsystem is a hydrazine fueled system which is combined with the orbit adjust subsystem. Common tankage is manifolded to 0.1 and 1.0 lb thrusters as well as the 5 lb. OAS thrusters. As stated above, the subsystem operates in a blowndown mode. The equipment is installed in a module mounted on the aft end of the S/C, see Figure D1.1.1-15.

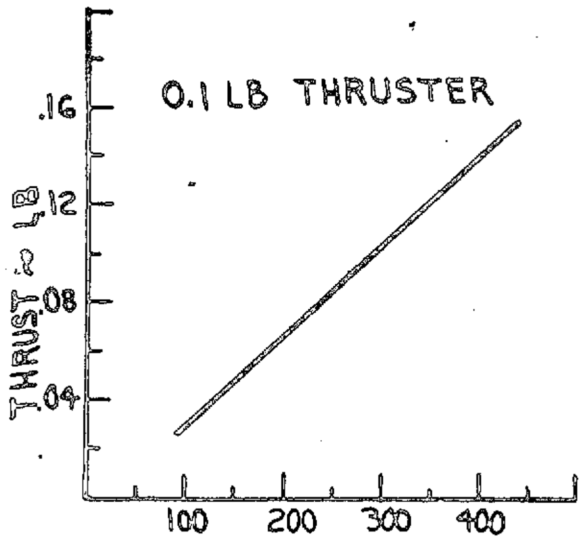
A single thread, minimum redundancy subsystem, shown schematically in Figure D 1.3.4-5, has been selected for the combined reaction control/orbit adjust subsystems. The majority of the components selected have been described above.

. The 0.1 lb thruster is currently being qualified for Flt SatCom. The 1.0 lb. thruster has been flown on Pioneer 10 and 11 and is being requalified for FltSatCom. Over the blowdown range of pressures the 0.1 lb. thruster has a thrust level range of 0.14 down to 0.04 lb, see Figure D1.3.4-6. The 1.0 lb thruster range is 1.1 to 0.43 lb, see Figure D 1.3.4-7.

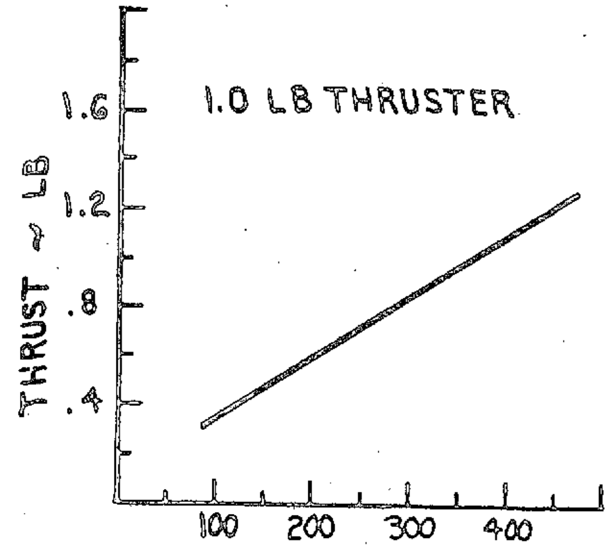
The subsystem selected has a minimum of redundancy. Referring to Figure D 1.3.4-5 it can be seen that the failure of a thruster in the on position can be sustained and the S/C can still be operated in a degraded mode. The failed-on thruster is shut-off with the latching solenoid valve. The remaining propellant in the associated tank can be used by opening the interconnect valve. The extent of degraded operation must be defined in the EOS preliminary design phase. As a minimum, the S/C can be operated in a survival mode until Shuttle retrieval or resupply is performed. Further redundancy to provide fail-safe operation can be added to the subsystem for minimal weight and cost. The addition of a second cross-over manifold and a second solenoid/seat assembly to each of the thruster valves, see Figure D 1.3.4-8, provides fail-safe operation. A total weight increase of 4.4 lb with a corresponding increase in cost of $20K provides fail-safe redundancy.

The selected reaction control/orbit adjust subsystem provides the flexibility to accommodate vehicle growth or mission changes.

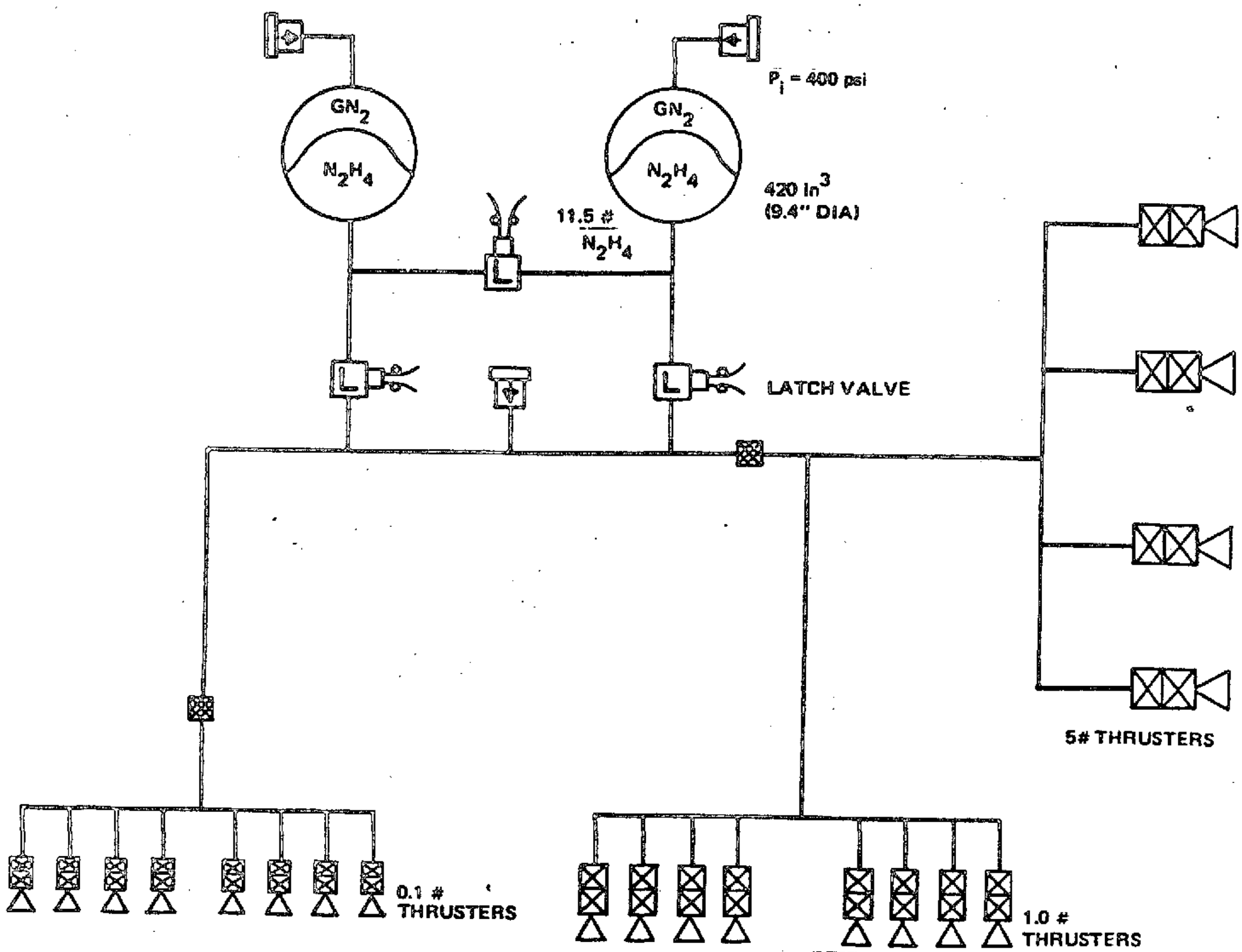
The structural arrangement of the RCS/OAS module is such that it is possible to mount and connect two additional tanks into the system. Thus, a total increase in propellant weight of 23 lb



THRUST VS TANK PRESS.
FIGURE D1.3.4-6



THRUST VS TANK PRESS
FIGURE D1.3.4-7



HYDRAZINE REACTION CONTROL/ORBIT ADJUST SUBSYSTEM
FAIL-SAFE REDUNDANCY
FIGURE D1.3.4-8

can be provided. It is well within the capability of the thruster designs to burn the additional propellant. Table D 1.3.4-3 shows that 80% of the total impulse (i.e. burned propellant) is supplied by the 5.0 lb thrusters. Assuming that on the average 2 of the 4 thrusters are firing, the 5 lb thruster burn time is increased by less than 500 seconds. Assuming the remainder of the propellant is split equally between the 0.1 and 1.0 lb thrusters and that only 4 of each out of 8 burn at any one time their burn times are increased ≈ 80 and ≈ 8 seconds respectively.

D 1.3.4.3 Orbit Transfer Subsystem

D 1.3.4.3.1 Requirements Definition

The Shuttle payload capability as defined by NASA, JSC establishes the requirement for an orbit transfer subsystem (OTS) or kick stage when the operational orbit exceeds $\approx$400 nm. Our studies selected an operational altitude of 366 nm eliminating the need for the OTS. However, two trade studies were conducted and the results are reported below.

D 1.3.4.3.2 Configuration Alternates

The primary means of providing orbit transfer capability is the use of solid rocket motors (SRM) as defined in the GSFC EOS System Concept Study (part of the RFP documentation). The alternatives studied were the orbit adjust subsystem using 75 lb thrusters (4) and a bipropellant system based on the Shuttle orbit maneuvering subsystem (OMS). For each case, it was assumed that the Shuttle would operate in a 300 nm orbit with the EOS being transferred to and from a 493 nm orbit.

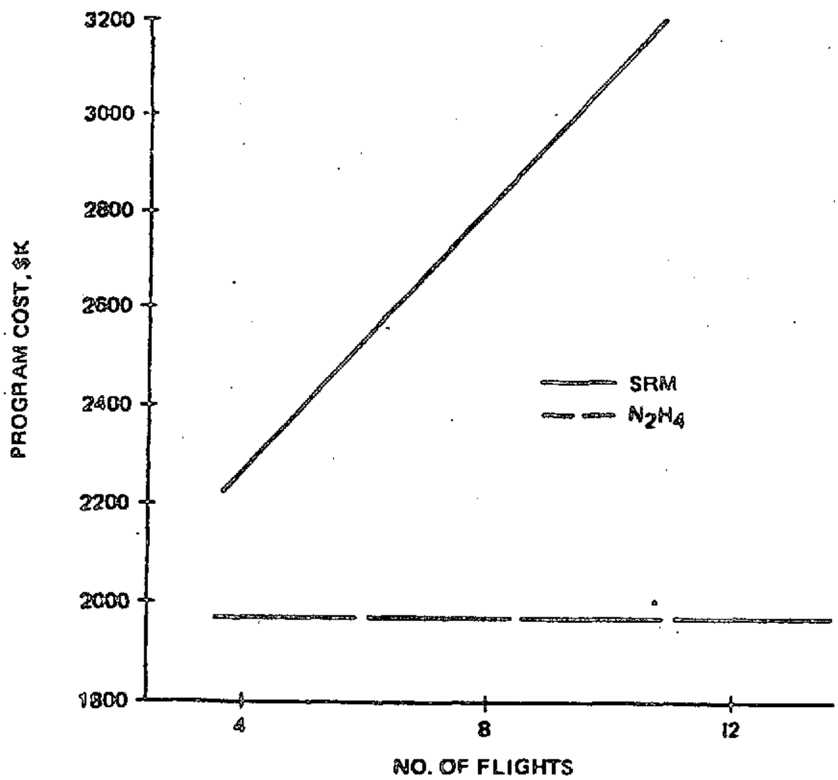
As stated above, the use of the OAS for orbit transfer required the replacement of the 5 lb with 75 lb thrusters. In addition, because of the much higher propellant load required, the two 9.4 inch tanks are replaced by three 22 inch tanks. With the exception of the 3 tanks the subsystem is schematically identical to the OAS/RCS shown in Figure D 1.3.4-5. Here again, in order to obtain an equal comparison, the combined SRM/OAS weight and costs were compared to the all N_2H_4 system. The results of the trade are shown in Table D 1.3.4-5.

	COMBINED SRM/OAS	ALL N_2H_4	DELTA (COMBINED vs ALL N_2H_4)
Weight, lb	405.6	464.6	+59
Cost, $K	2,283	1,977	-306

Table D 1.3.4-5. SRM vs N_2H_4 Orbit Transfer Subsystem

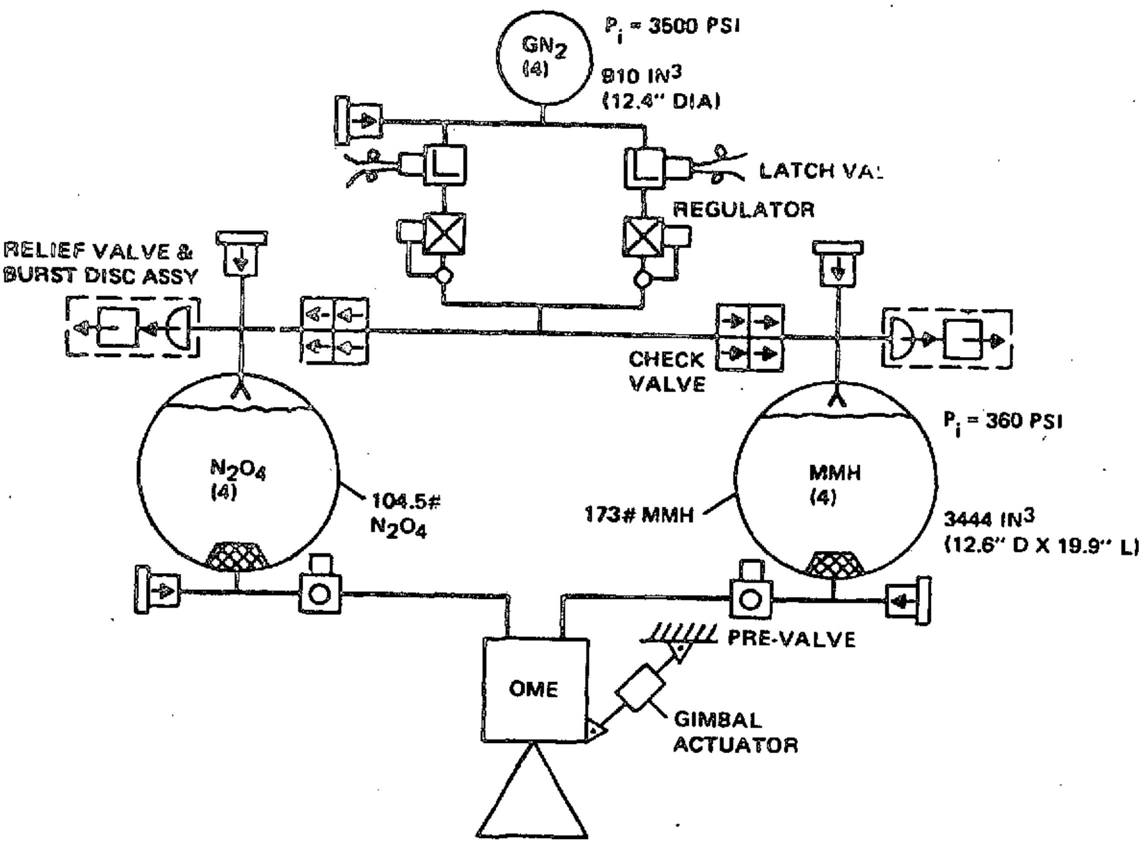
It should be noted that the above costs are based on a four vehicle/four flight program. As the number of flights increases, the cost differential becomes extremely large. At 12 flights, the cost differential exceeds $1M, see Figure D 1.3.4-9.

The use of a bipropellant orbit transfer subsystem appears to be viable only for the larger EOS spacecraft being studied; vehicles which require orbit transfer stages such as the Boeing Burner II type design. This study assumed the use of the SVM-2 motors called for in the Boeing design. A bipropellant system using N_2O_4 and MMH and sized to the same total impulse as the 4 SVM-2's was assumed. The system is shown schematically in Figure D 1.3.4-10. A four vehicle/four flight program was also assumed. The results are shown in Table D 1.3.4-6.



PROGRAM COST SAVINGS

FIGURE D.1.3.4-9



ORBIT TRANSFER SUBSYSTEM (LIQUID ENGINE ~ OME)

FIGURE D.1.3.4-10

	SRM's	BIPROPELLANT	DELTA (SRM vs BIPROP)
Weight, Lb	1400.0	1477.1	+ 77.1
Cost, $K	1,247	2,705	+ 1,458

Table D 1.3.4-6. SRM vs Bipropellant Orbit Transfer Subsystem

At first glance, the bipropellant system appears to be a poor choice. However, this system uses Shuttle hardware which is designed to operate for 100 missions. It is, therefore, capable of operating over the full lifetime of the EOS. Figure D 1.3.4-11 shows that a cross-over point occurs in total program costs at the 10-11 flight point in the program.

D 1.3.4.3.3 Selected Configurations

The results of the N_2H_4 fueled OTS study reflects a cost savings of $306K at a weight penalty of 59 lb. Should later studies show operation at altitudes above 400 nm are required, the N_2H_4 OTS should be considered for Delta or weight constrained Titan III B spacecraft designs.

The results of the bipropellant fueled OTS study shows the system to be too costly for programs with less than 11 flights. For Titan III B spacecraft designs with more than 11 flights, the bipropellant system is a viable though more complex approach to the OTS design.

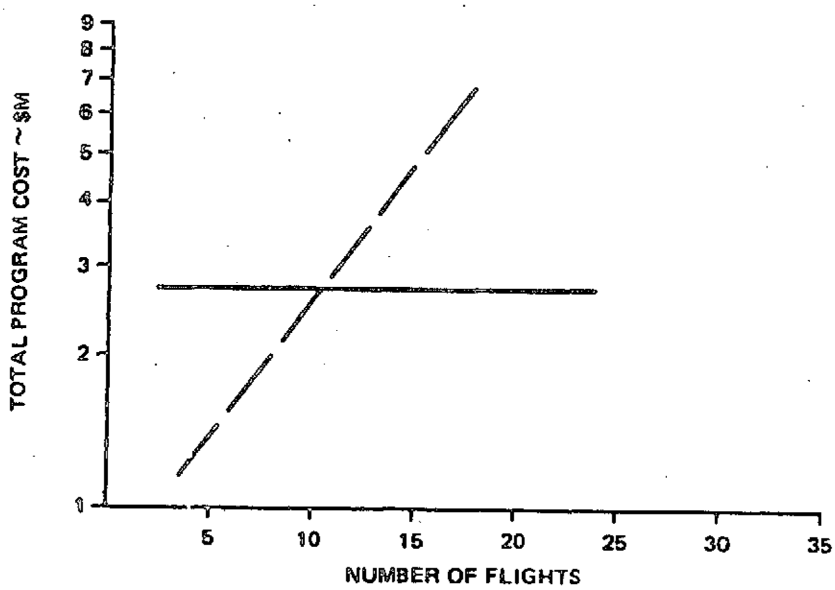


Fig. D.1.3.4-11 SRM vs Bipropellant OTS Total Program Costs

D 1.3.4.4 Candidate Hardware

The hardware (components) considered for incorporation into the subsystems discussed above are listed in Table D 1.3.4-7. The selected components used in our trade studies have been marked with a ✓.

Table D 1.3.4-7 Candidate Components

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL.	PROCUREMENT COST, $K		REMARKS
PART NAME	SELLER	P/N	STATUS	FUNCTION/ CAPABILITY	WT.(LB)	AVG. PWR.	STATUS	NON-RECUR	UNIT COST	
Propellant Tank (N_2H_4) (10" Dia)	Pressure Systems Inc.	80156-1 (AEROS)	Exists	Vol = 420in^3, Oper.Press=610psi Diaphragm expulsion	2.91	-	Similarity	10	13	
	Airite	-	New			-	Not. Qual.	42.4	9.7	
	TRW	413535	Exists	Vol = 465in^3, Oper.Press=500psi Bladder expulsion	3.75	-	Similarity	8	12.5	
Propellant Tank (N_2H_4) (16.5"Dia)	Pressure Systems, Inc.	80177-1 (ATS)	Exists	Vol.=2300in^3, Oper.Press=400psi Diaphragm expulsion	10.8	-	Similarity	10	19	
	Airite	- 9183	Exists	Vol=2300in^3, Oper.Press=500psi Diaphragm expulsion	11.0	-	Δ Qual.	55.7	13.6	
	Ardo	E3848 (USAF/LMSC)	Exists	Vol=2625in^3, Oper.Press=350psi	18.8	-	Δ Qual.	18 +	19.6	
Propellant Tank (N_2H_4)(22" Dia)	Pressure Systems, Inc.	80112-115 (P-95)	Exists	Vol =5580in^3, Oper.Press=350psi Diaphragm expulsion	17.25	-	Similarity	10	22	
	Airite	-	New			-	Not Qual.	68.8	17.3	
Propellant Tank (GN_2) (12" Dia)	Pressure Systems, Inc.	80135-1 (Atlas/Centaur)	Exists	Vol = 886in^3, Oper. Press=3400psi	15.0	-	Similarity	6	4.7	
	Airite	6499-5	Exists	Vol = 910in^3, Oper.Press=2600psi	8.3	-	Similarity	2.6	6.9	
Propellant Tank (GN_2) (≈16" Dia)	Pressure Systems, Inc.	80088-1	Exists	Vol =1728in^3, Oper.Press=3600psi	21.8	-	Similarity	6	6.8	
	Airite	6384	Exists	Vol =2200in^3, Oper. Press=2800psi	21.6	-	Similarity	3	10.1	
	Ardo	E3749	Exists	Vol =1960in^3, Oper. Press=3750psi	27.5	-	Δ Qual	17.5 +	7.5	
Propellant Tank (GN_2) (≈20" Dia)	Pressure Systems, Inc.	80097-1 (J28/Saturn)	Exists	Vol = 4000in^3 Oper.Press=4000psi	63.4	-	Similarity	6	8	
	Airite	6331	Exists	Vol =4650in^3, Oper Press=2500psi	50.0	-	Similarity	3.5	12.9	

Sheet 1 of 3

Table D 1.3.4-7 Candidate Components (cont)

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL. STATUS	PROCUREMENT COST, $K		REMARKS
PART NAME	SELLER	P/N	STATUS	FUNCTION/CAPABILITY	WT(lb)	AVG PWR.		NON-RECUR	UNIT COSTS	
Thruster(N_2H_4) (75#)	Rocket Research	-	New	Isp=230sec, ε=50, Pin=350psi	5.54	-	Not Qual	239.4	20.7	✓
	Ham. Std.	-	Mod.Exist.	Isp=238 sec, - , Pin=300psi	5.5	-	ΔQual.	350	25	
	Bell Aerospace	-	New				Not Qual	371	21	
	Marquardt	R-30-1 (USAF Set)	Mod.Exist.		3.3		Similarity	50	21	
Thruster(N_2H_4) (5#)	Rocket Research	MR-50	Exists	Isp=231, sec ε=40, Pin=350psi	1.2		Similarity	32.1	11.3	
	Ham. Std.	REA16-7	Exists	Isp=235, , Pin=260psi	0.75		Similarity	-	15	✓
	Bell Aerospace	8717	New		0.74		Not Qual	203	12	
Thruster(N_2H_4) (1#)	TRW	410618 (FLT SATCOM)	Exists	Isp=226 sec, - Pin=350psi	0.7		Similarity	25	12	✓
Thruster(N_2H_4) (0.1 #)	TRW	MRE-1/10 (FLT SATCOM)	IN Qual	Isp=221 sec , Pin=300psi	0.50		Similarity	25	6.2	✓
	HAM STD.	REA 10-15 (CTS)	Exists	Isp=227 , Pin=300psi	0.34		Similarity	25	11.7	
	Rocket Research	MR-74 (ATS F & G)	Exists	Isp=221 sec, Pin=300psi	0.51		Similarity	25	10.6	
Thruster(GN_2) (0.1#)	Fairchild	683000 (OAO)	Exists	Pin=40psi		25	ΔQual	7.8	1.7	
	Sterer	51350 (Nimbus, P72-2)	Exists	Pin=40psi		19	Similarity	–	5.6	
	Wright Components	15751 (ELMS)	Exists	Isp=67 sec, ε=20, Pin=42psi	0.5		Similarity	0.7	1.1	✓
Thruster(GN_2) (1.0#)	Walter Kidde	872458 (Atlas Nose Cone)	Exists	Pin=3250psi		34	Similarity	–	3.4	
	Valcor	V-27200-288 (ELMS)	Exists	Isp=70.5sec ε=20, Pin=2500psi	0.51	37	Similarity	-	4	✓
	Sterer	31980 (Polaris)	Exists	Pin=2000psi		37	ΔQual	–	1	Redesign for higher press.

Sheet 2 of 3

Table D 1.3.4-7 Candidate Components (cont)

CANDIDATE COMPONENTS				SIGNIFICANT TECHNICAL FEATURES			QUAL. STATUS	PROCUREMENT COST, $K		REMARKS
PART NAME	SELLER	P/N	STATUS	FUNCTION/CAPABILITY	WT (lb)	AVG. PWR.		NON-RECUR	UNIT COST	
Regulator (GN_2)	Fairchild Sterer	60100 (OAO) 53800 (ELMS)	Exists Exists	W=0.001 #/sec, Pout=30-55psia W=0.0034 #/sec, Pout=50±3psia	1.29	—	Δ Qual. Similarity	33 1	7.2 12	✓ Dual seat with relief valve.
Filter (N_2H_4 & GN_2)	Vacco	F1D10064-01 (Intelsat IV)	Exists	10 μ absolute	0.3	—	Similarity	—	1	✓
Fill Disconnect (N_2H_4 & GN_2)	Pyronetics	1831-6 & 7 (Viking Lander)	Exists		0.22	—	Similarity	1.1	0.6	✓
Latching Solenoid Valve (N_2H_4)	Carleton Controls	8217001-2 (Intelsat IV)	Exists		0.61		Similarity	—	3.5	✓
Latching Solenoid Valve (GN_2)	Valcor	V27200-454 (ELMS)	Exists	W=.0077 #/sec @ 218 psia, Pmax = 3620 psia	0.65	30	Similarity	—	4	✓
Solid Rocket Motors	Thiokol Aerojet	TE-M-516 SVM-8	Mod. Exist Exists	ISP=286 sec, I_t = 82,250#sec Isp=281 sec, I_t=86,900 #sec	88 350		Δ Qual Δ Qual	883 76.7	55 76.7	✓ ✓

1.3.5 Subsystem Thermal Control

1.3.5.1 Summary

The thermal evaluation of the subsystems was based on a modular configuration. Two module configurations were considered for the Delta triangular arrangement and a square configuration was considered for a Titan arrangement.

Evaluations were conducted for the Land Resources Mission. Worst case min/max environment heat fluxes were used for each module (see section 1.2.3). An altitude range of 300 nm to 500 nm and DNTD range of 9:30a.m. to 12:00 Noon was used as the basis for determing the worst case heat fluxes. Where applicable, heat input from the solar array was also included.

Up to this point in the study the subsystems were treated on a cost/capability trade-off basis. The matrix of black box combinations within a subsystem has made it virtually impossible to evolve a load analysis of meaningful value. In addition, module layouts have not been generated. Therefore the analysis of the modules could only be considered on a lumped parameter, parametric basis. The ability to reject heat was studied as a function of alternate thermal options for each location. This technique established module location and feasibility of passive control, supplemented with heater power during low power dissipating modes.

Section 1.2.2 showed the cost per watt can vary between .75K and 1.75K, depending on the array selected. The savings in module acceptance test costs resulting from a narrow operating temperature range ($\pm$ 10°F vs. $\pm$ 50°F) can be as much as 16K. The fundamental passive design cost trade-off is therefore the impact of equipment operating temperature range on power subsystem and test costs. The cost of active control to reduce heater power(if a true penalty)must then be considered. These tradeoffs are used to achieve the design-to-cost targets.

1.3.5.2 Module Thermal Evaluation General

The configurations evaluated are shown in Fig. 1.2-1, and have been designated Delta 1, Delta 2, and Titan. Delta 1 is the selected module configuration and Delta 2 was an alternate arrangement which was considered.

The modules were assumed to have a heat sink effective radiating area equal to 15.6 $ft.^2$. The other five surfaces of each module were insulated. The modules and structure were considered to be at the same temperature (consistent with the module support structure evaluation).

Figures 1.3.5-1 to 1.3.5-16 presents the heat rejection capability of each module considered for hot and cold cases as a function of beta(β). The term beta is the radiation coupling between the module heat sink and space and encompasses all possible thermal design approaches, including module heat sink emittance presence or absence of skins, and insertion of louvers or heat pipes. A value of beta equal to .40 is indicative of an all-black heat sink, skin-on design. A beta value of .80 represents an all black heat sink with no skin. Table 1.3.5-1 summarizes the hot case heat rejection capability for all module locations.

The determination of subsystem heater power penalty is not as straight forward as with the structure. A system approach must be used to evaluate true heater power penalties. The power subsystem must be sized to provide power for the other subsystems. Heater power should not be considered a penalty as long as the total of power dissipation and heater power does not exceed the total power used for sizing the power subsystem. The design of the power subsystem will determine whether the power dissipation of this subsystem varies directly or inversely with total system power. If the power dissipation varied inversely with total system power a heater power penalty may result. With the above heater power philosophy as a basis, passive techniques in most cases will yield acceptable designs.

1.3.5.3 EPS Module

The EPS modules have been located on the Delta and Titan configurations to provide good heat rejection capability. For the Delta 1 configuration, the hot case maximum heat rejection at 70°F is 170 watts (10.9 watts/ft^2) for a skin-on type design and 340 watts (21.8 watts/ft^2) with no skin. The Delta 2 configuration has a slightly higher hot case heat rejection capability, however a larger variation in heat rejection from the hot to cold case results in heater power penalties. Therefore the Delta 1 configuration has been selected. The Titan location provides the maximum heat rejection for that configuration of modules. The hot and cold case heat rejection capabilities are shown in Figures 1.3.5-1,2, 7,8,13 and 14 and the maximum hot case heat rejection at 70°F is given in Table 1.3.5-1. All hot and cold cases include effects of orbital heat fluxes and theromphysical property changes.

On a total module basis maintaining 70°F ± 20°F yields a heat rejection ratio of 1.36 (for Delta 1, hot to cold case, skin-on design). As long as the power dissipation variation does not exceed this ratio, no heater power is required.

The battery area of the module must be sized to accommodate 50 watts at 50°F and 25 watts at 30°F. A split heat sink will be considered to separate the batteries. The Delta 1 configuration at 50°F has a heat rejection capability of 8.3 watts/ft^2 (skin-on) to 16.6 watts/ft^2 (skin-off). A skin-on design therefore requires 6. FT^2 of heat sink to reject the heat. The cold case heat rejection at 30°F (skin-on) is 7.6 watts/ft^2. A 6. FT^2 heat sink would reject 46 watts, thereby causing 21 watts of heater power. A skin-off design causes even larger heater power. Thus, from both packaging and heater power viewpoints, the use of active control, in the form of variable conductance heat pipes, is the most favorable cost effective solution. Heat rejection capability approaching that of a skin-off design is achieved, heater power is reduced, and operating temperature range is reduced.

1.3.5.4 ACS Module

The ACS module heat rejection capabilities are given in Figures 1.3.5-3, 1.3 1.3.5-9, 1.3.5-10. The Delta 1 and Titan are the same location providing good heat rejection and a favorable anti-earth location for the subsystem. The Delta 2 configuration gives approximately 28% less heat rejection capability. Preliminary power dissipation information indicates that a passive technique can be used to achieve a narrower operating temperature range of 70°F ± 10°F to 70°F ±20°F with no true heater power penalties.

1.3.5.5 C&DH Module

The C&DH module heat rejection capabilities are given in Figures 1.3.5-5 1.3.5-6, 11, 12, 15 & 16. Based upon lower anticipated heat rejection requirements this module has been located on the solar array side of the structure for each configuration. The heat rejection capabilities for each Delta configuration are approximately the same, therefore the Delta 1 configuration was selected. Passive techniques will be sufficient for maintaining a narrow operating temperature range (i.e., 70°F ± 20°F) with no significant true ~~heater~~ power penalties

This can be seen on Figure 1.3.5-5 and 1.3.5-6 where the heat rejection capability at 90°F equals the heat rejection at 50°F.

The heat rejection of the Titan module is severely limited by the heat flux from the solar array. There is virtually no heat rejection capability at 70°F for a skin-on design. Several alternatives are available including 1) removal of the skin and using a stable coating, 2) selecting a higher operating temperature or 3) locating the module on the earth-facing side (Fig.1.3.5,9same as ACS Delta 2 location). When the load requirements have been established, the alternatives will be evaluated and traded against each other.

1.3.5.6 Future Efforts

The next phase of the study will concentrate on a more detailed evaluation. With a load analyses, module layouts and refined flexible array costs, the final trade-off can be made to achieve the design-to-cost targets and design details developed.

TITLE	TRADE STUDY REPORT NO.
SUBSYSTEM THERMAL CONTROL	WBS NUMBER

TABLE 1.3.5-1

SUMMARY OF SUBSYSTEM MODULE HOT CASE
MAXIMUM HEAT REJECTION CAPABILITY AT 70°F

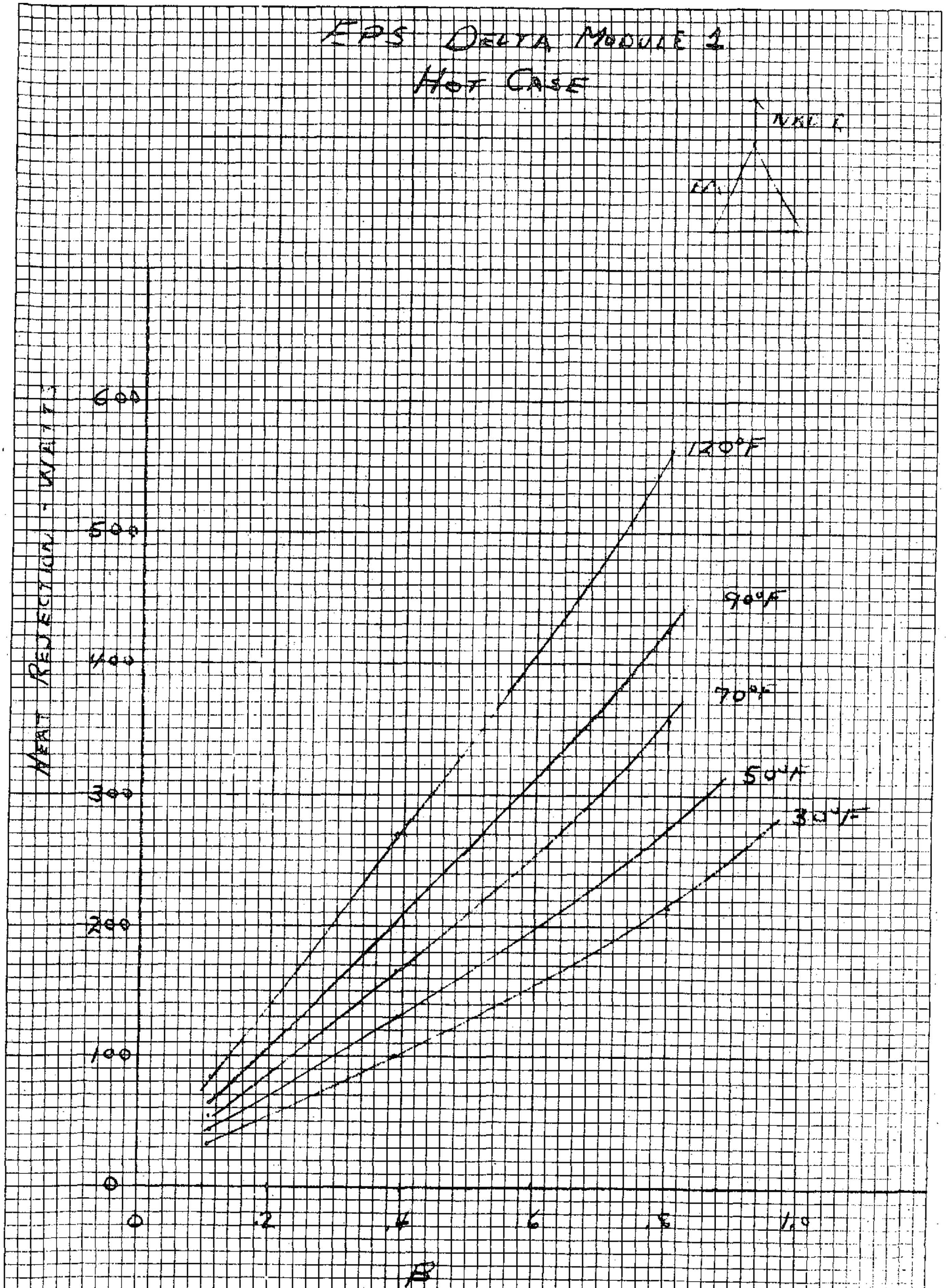
MODULE	SKIN ON WATTS (1)	SKIN OFF WATTS (2)
EPS Delta 1	170	340
EPS Delta 2	200	400
EPS TITAN	200	400
ACS Delta 1	180	360
ACS Delta 2	130	260
ACS Titan	180	360
C&DH Delta 1	90	180
C&DH Delta 2	110	220
C&DH Titan	2	90* (3)
C&DH Titan (alt)	110	220

(1) Heat sink emittance = .89 skin interior emittance = .89, skin external emittance = .76 skin external absortance = .18

(2) Heat sink emittance = .76, solar absortance = .18 except when noted

(3) Heat sink emittance = .80, solar absortance = .12

PREPARED BY	GROUP NUMBER & NAME	DATE	CHANGE LETTER
			REVISION DATE
APPROVED BY			PAGE OF



EPS DELTA MODULE 1
HOT CASE
HEAT REJECTION - WATTS
600
500
400
300
200
100
0
0
.2
.4
.6
.8
1.0
β
120°F
90°F
70°F
50°F
30°F

FIGURE 1.3.5-1

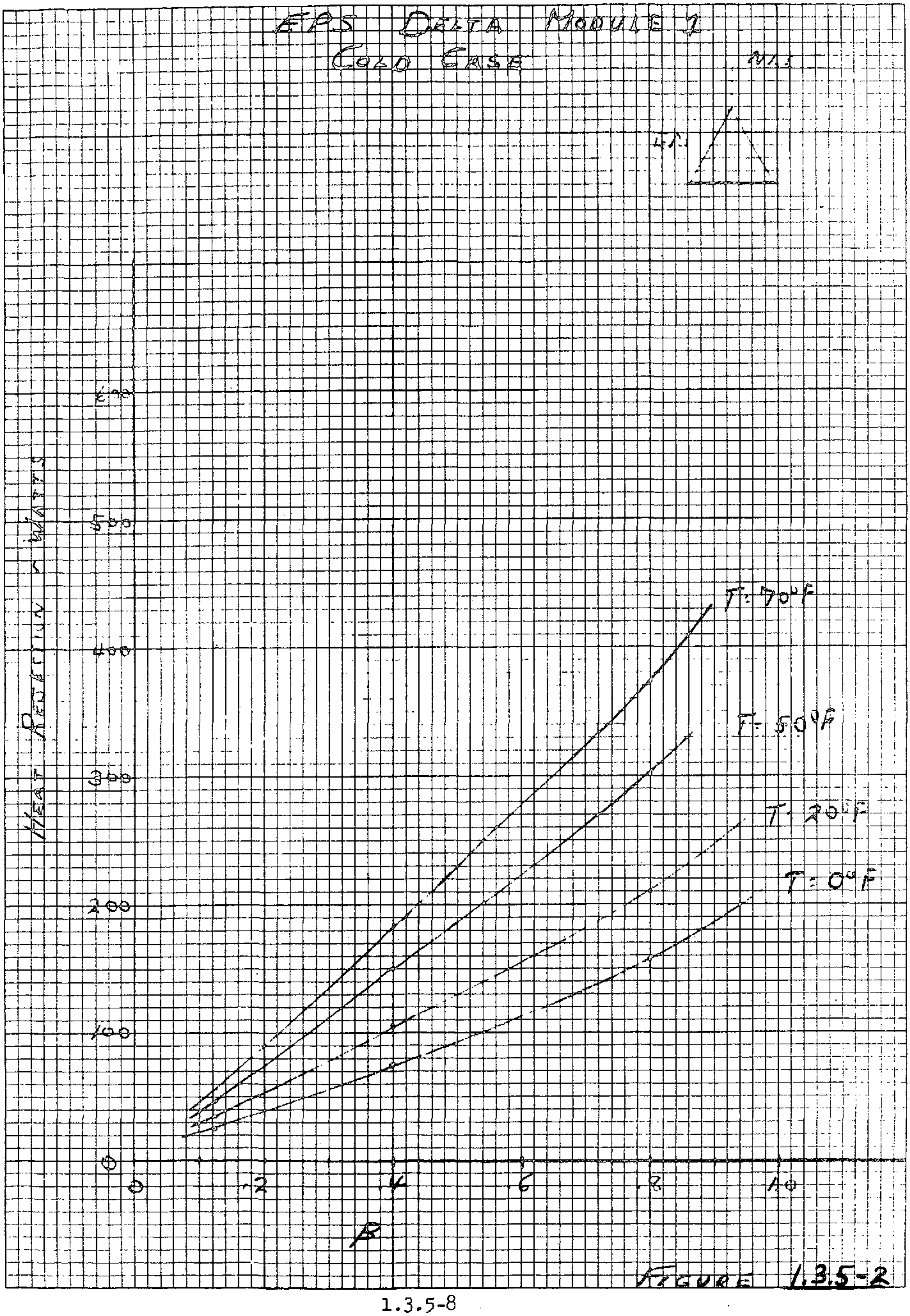
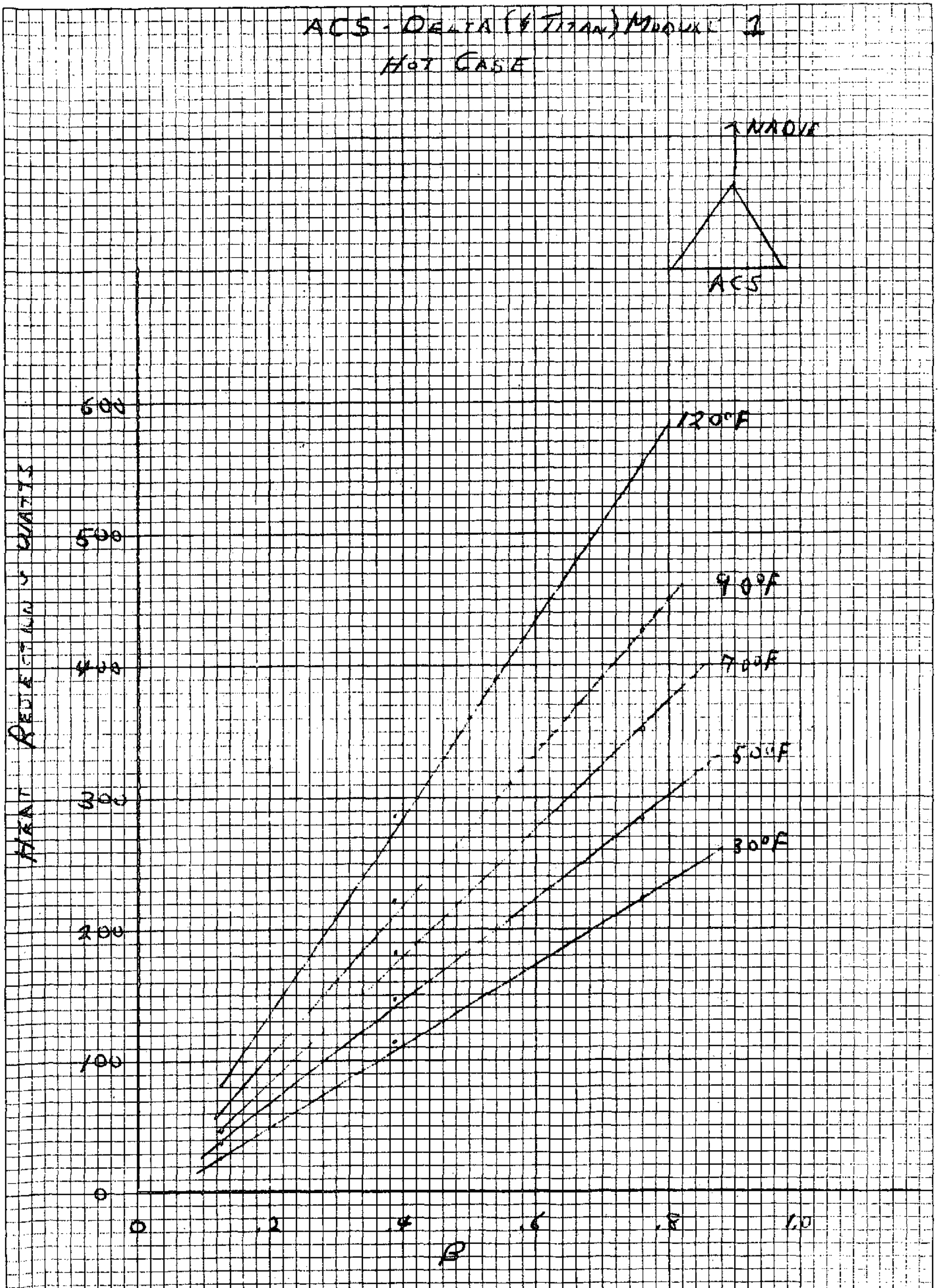

EPS DELTA MODULE 1
COLD CASE
HEAT REJECTION ~ WATTS
β
T = 70°F
T = 50°F
T = 20°F
T = 0°F
0
100
200
300
400
500
600
0
.2
.4
.6
.8
1.0

FIGURE 1.3.5-2



ACS - DELTA (& TITAN) MODULE 1
HOT CASE
NADIR
ACS
120°F
90°F
70°F
50°F
30°F
HEAT REJECTION ~ WATTS
600
500
400
300
200
100
0
0
.2
.4
.6
.8
1.0
β

FIGURE 1.3.5-3

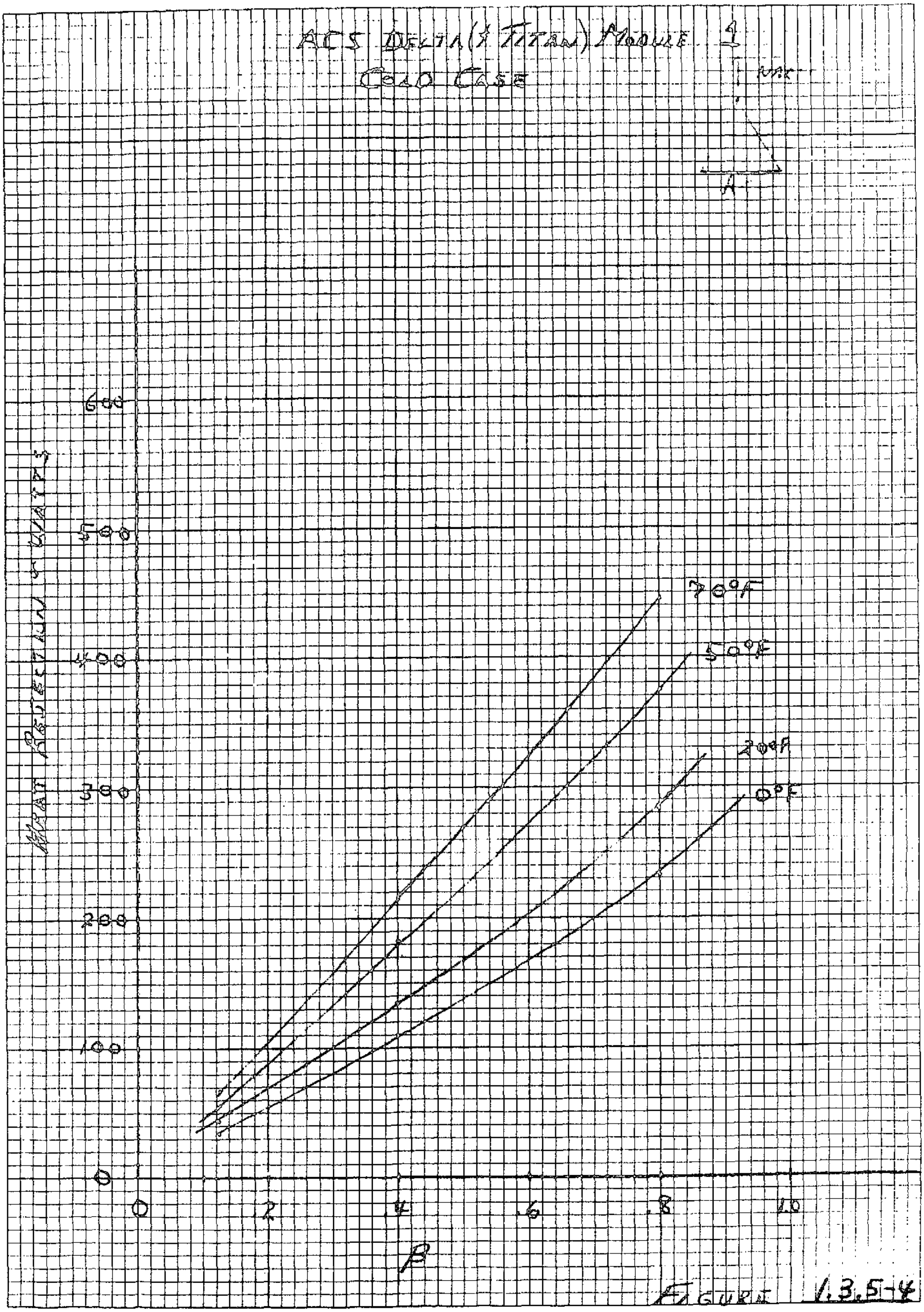


FIGURE 1.3.5-4

KEUFFEL & ESSER CO. MADE IN U.S.A.

46 0703

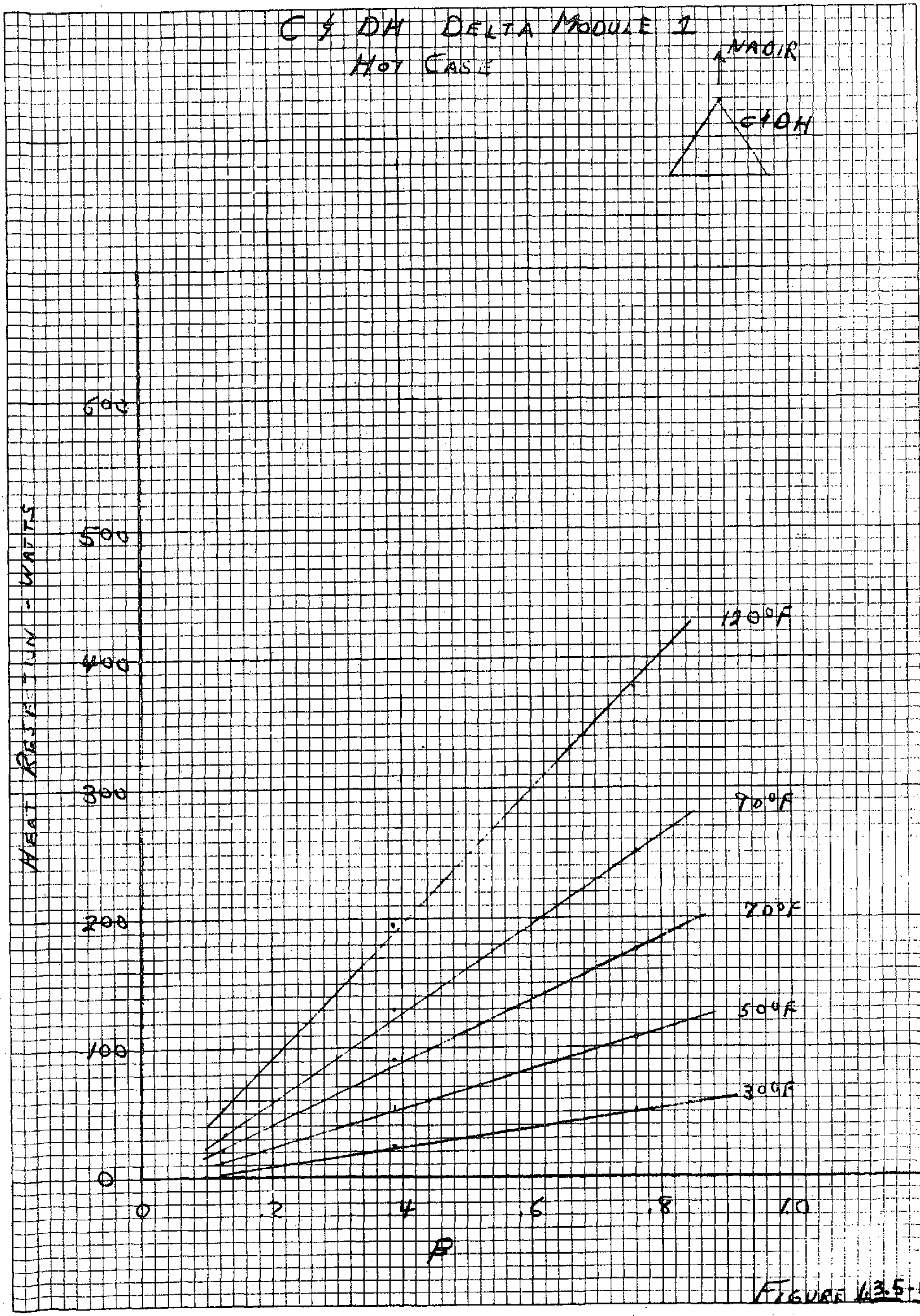

C & DH DELTA MODULE 1
HOT CASE
NADIR
C & DH
HEAT REJECTION - WATTS
600
500
400
300
200
100
0
120°F
90°F
70°F
50°F
30°F
0
.2
.4
.6
.8
1.0
β

FIGURE 1.3.5-5

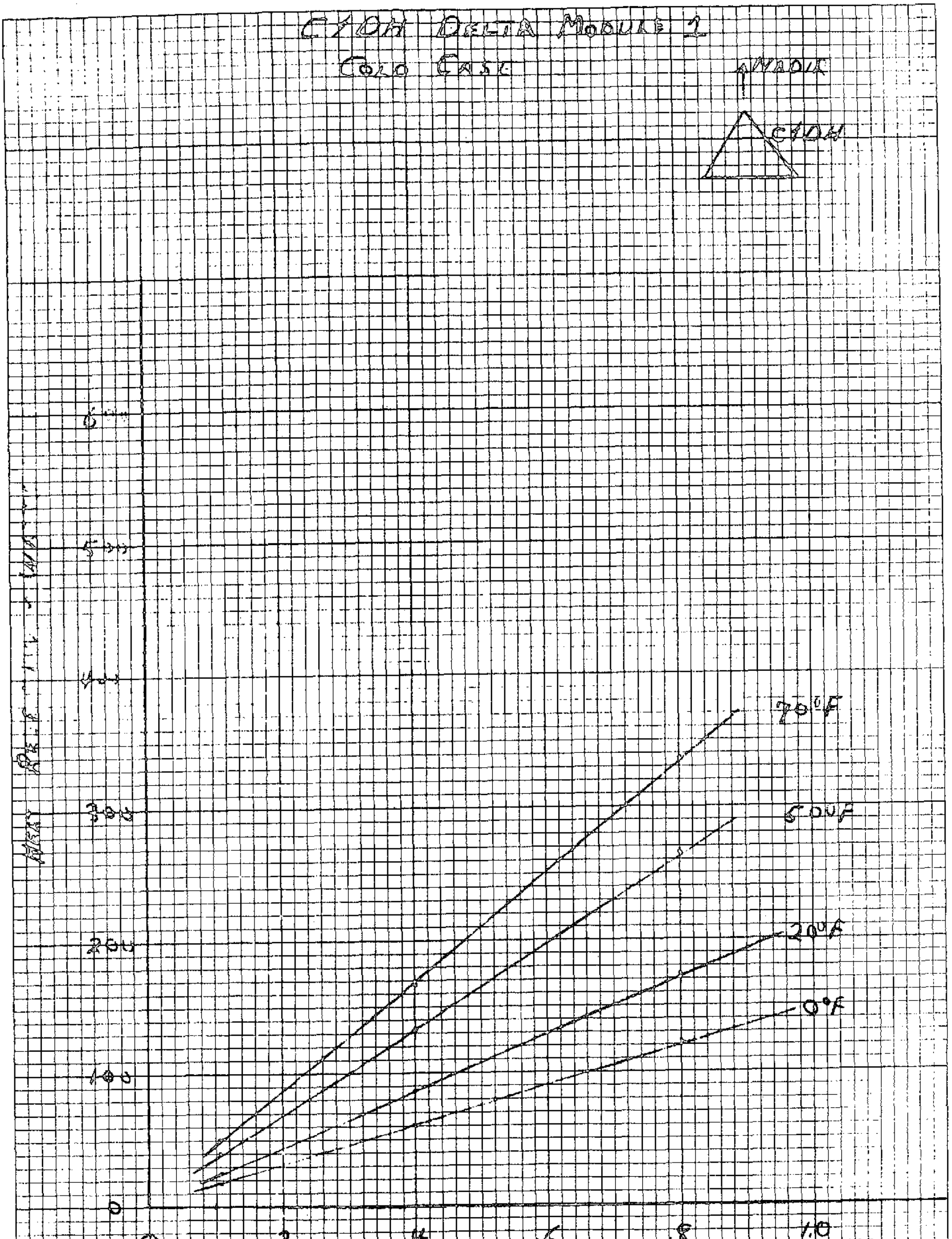
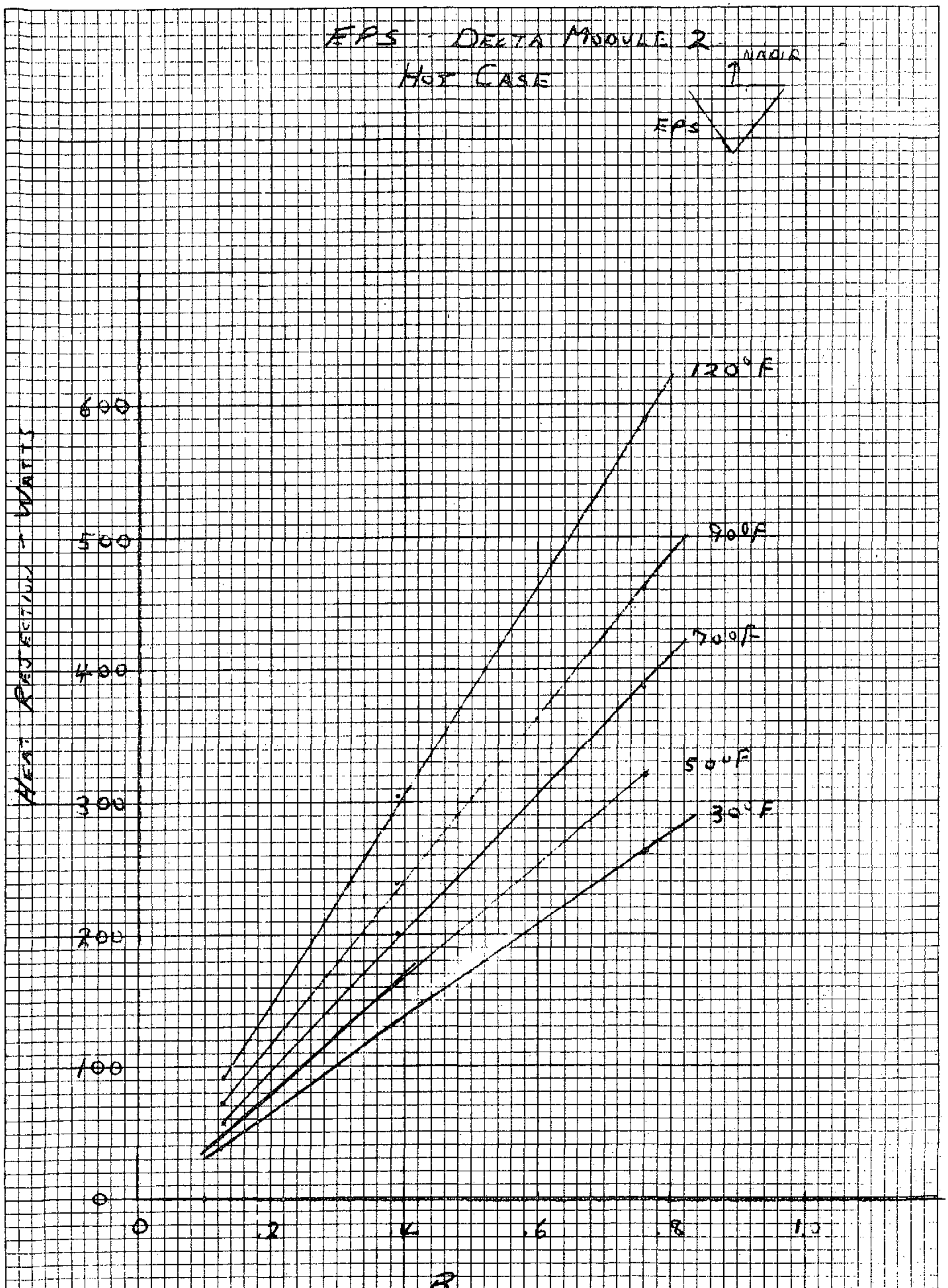
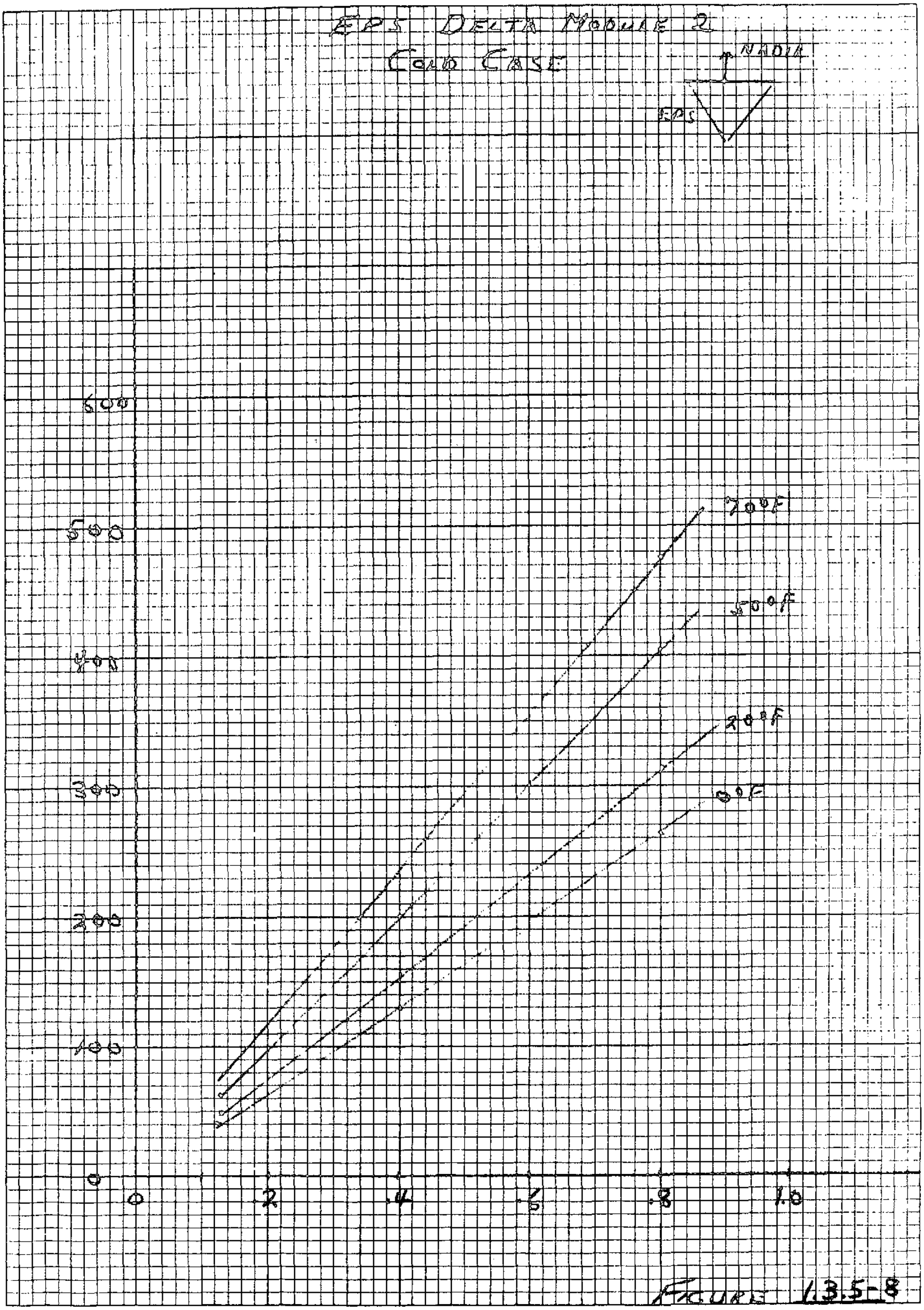

C1OH DELTA MODULE 1
COLD CASE
NADIR
C1OH
70°F
60°F
20°F
0°F
0
.2
.4
.6
.8
1.0
100
200
300
400
500
600
ρ

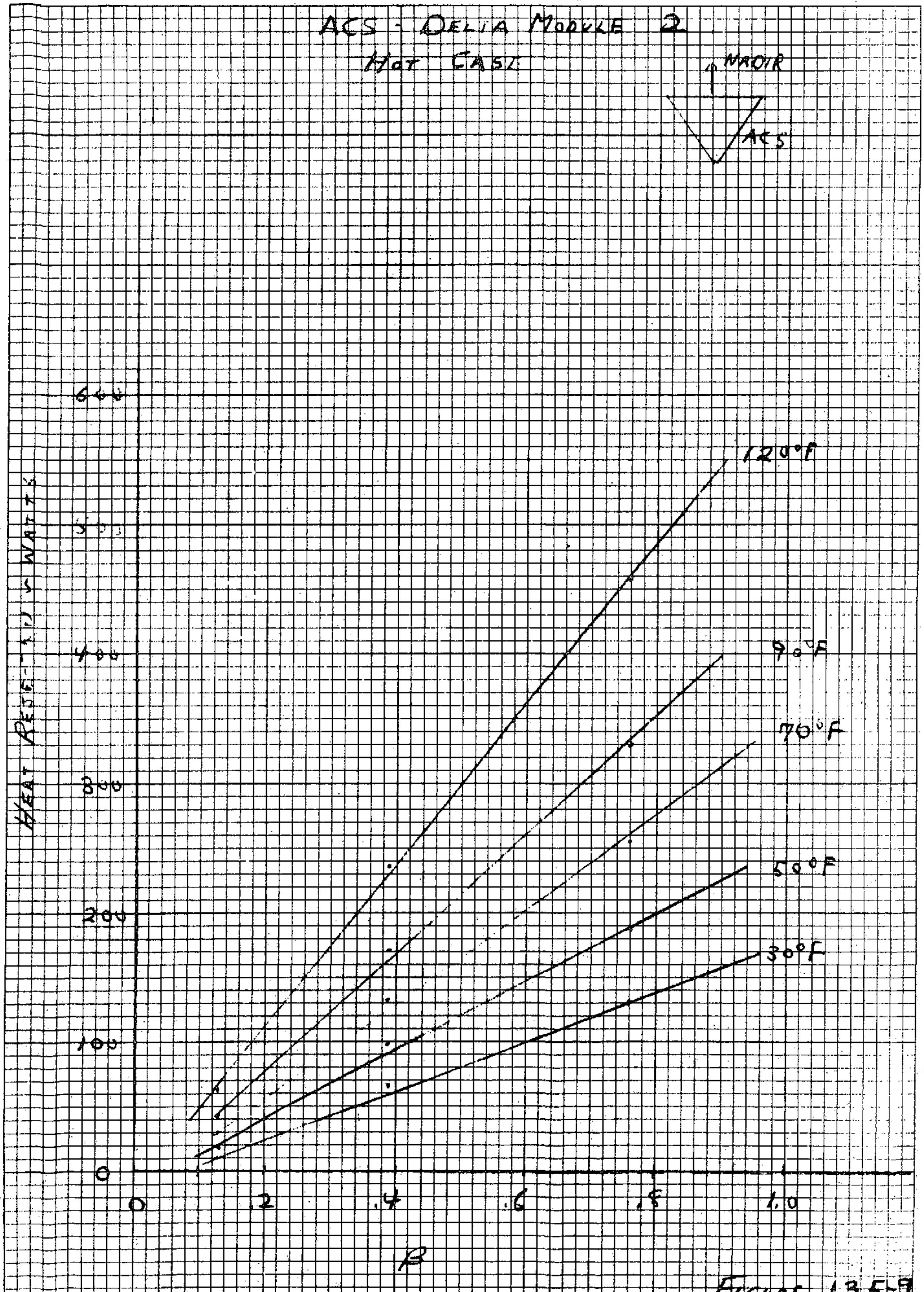
FIGURE 1.3.5-6



EPS DELTA MODULE 2
HOT CASE
NADIR
EPS
120°F
90°F
70°F
50°F
30°F
600
500
400
300
200
100
0
HEAT REJECTION - WATTS
0
.2
.4
.6
.8
1.0
β

FIGURE 1.3.5-7


EPS DELTA MODULE 2
COLD CASE
NADIR
EPS
700°F
500°F
200°F
0°F
600
500
400
300
200
100
0
0
.2
.4
.6
.8
1.0
FIGURE 1.3.5-8


ACS - DELTA MODULE 2
HOT CASE
NADIR
ACS
120°F
90°F
70°F
50°F
30°F
600
400
300
200
100
0
0
.2
.4
.6
.8
1.0
β
HEAT REJE- ... - WATTS
FIGURE 1.3.5-9

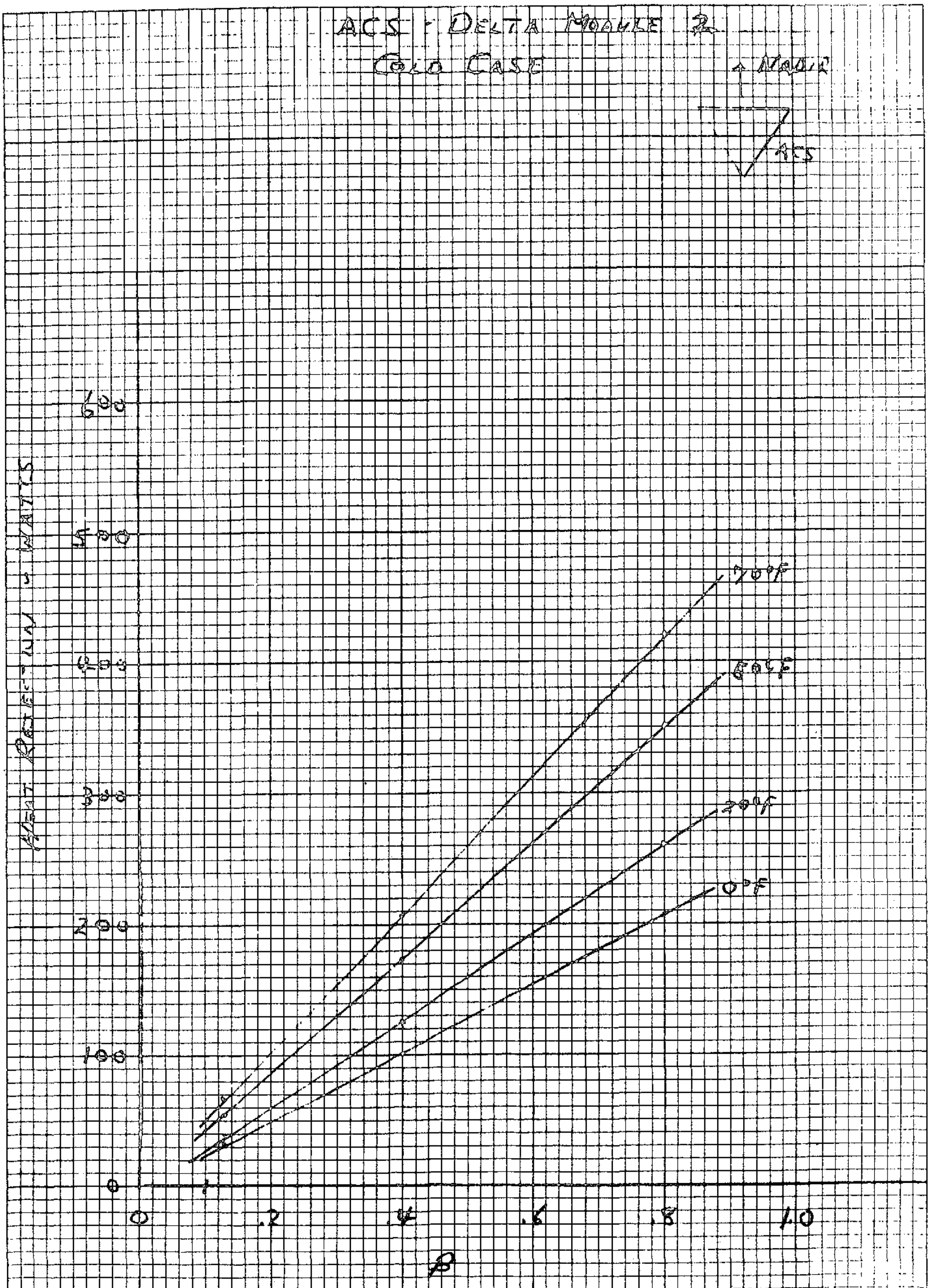


FIGURE 1.3.5-10

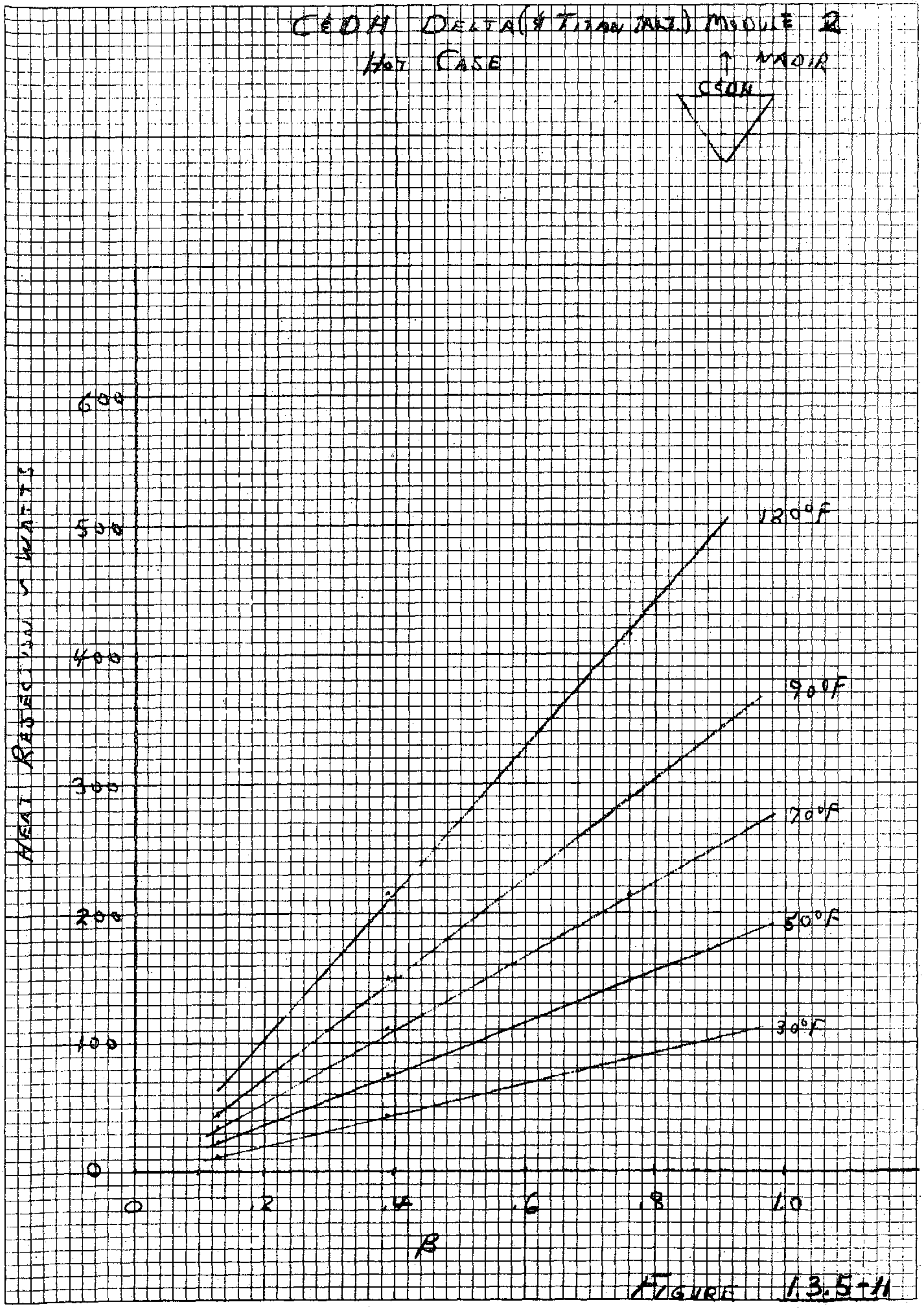


FIGURE 1.3.5-11

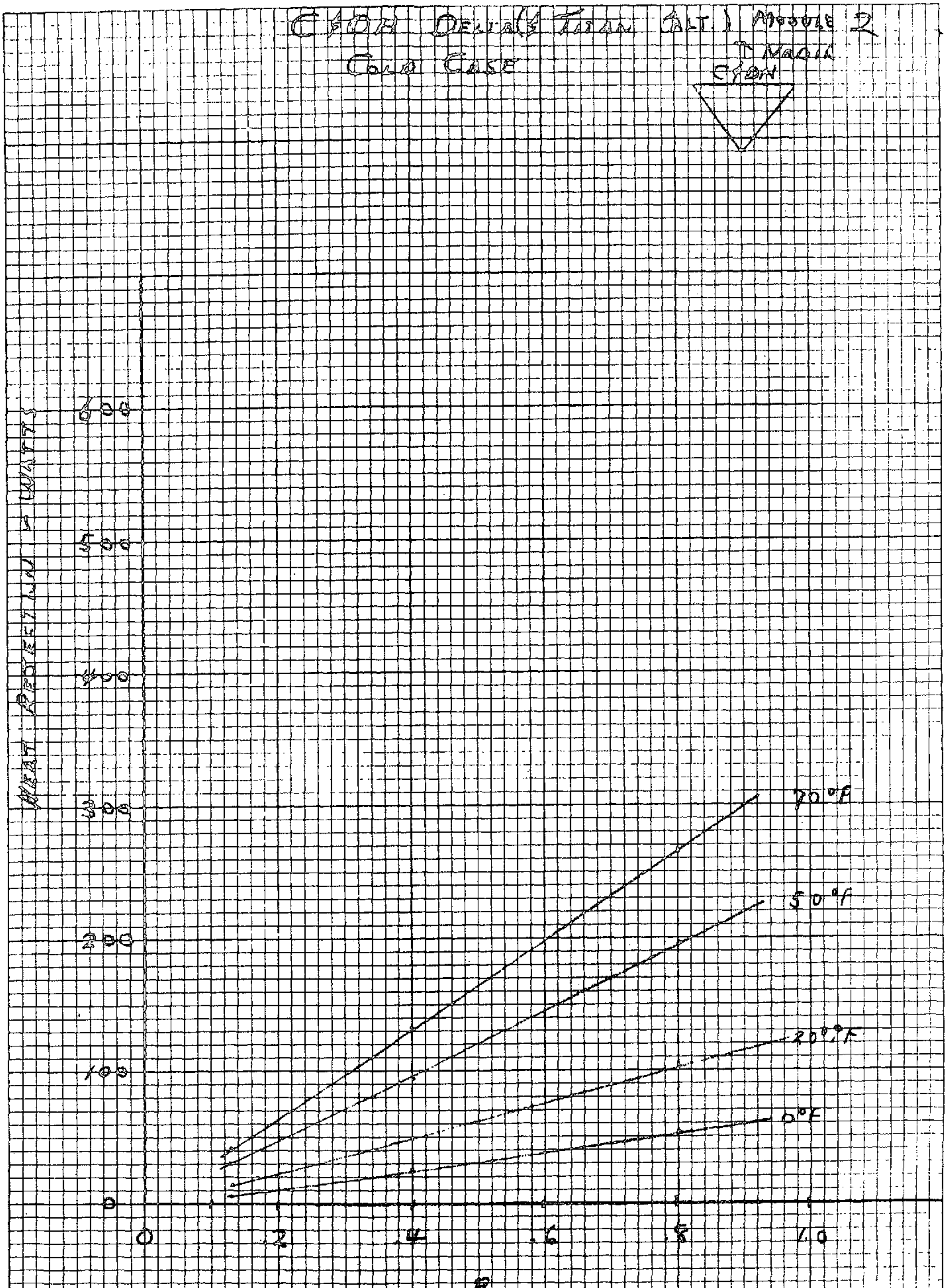


FIGURE 1.3.5-12

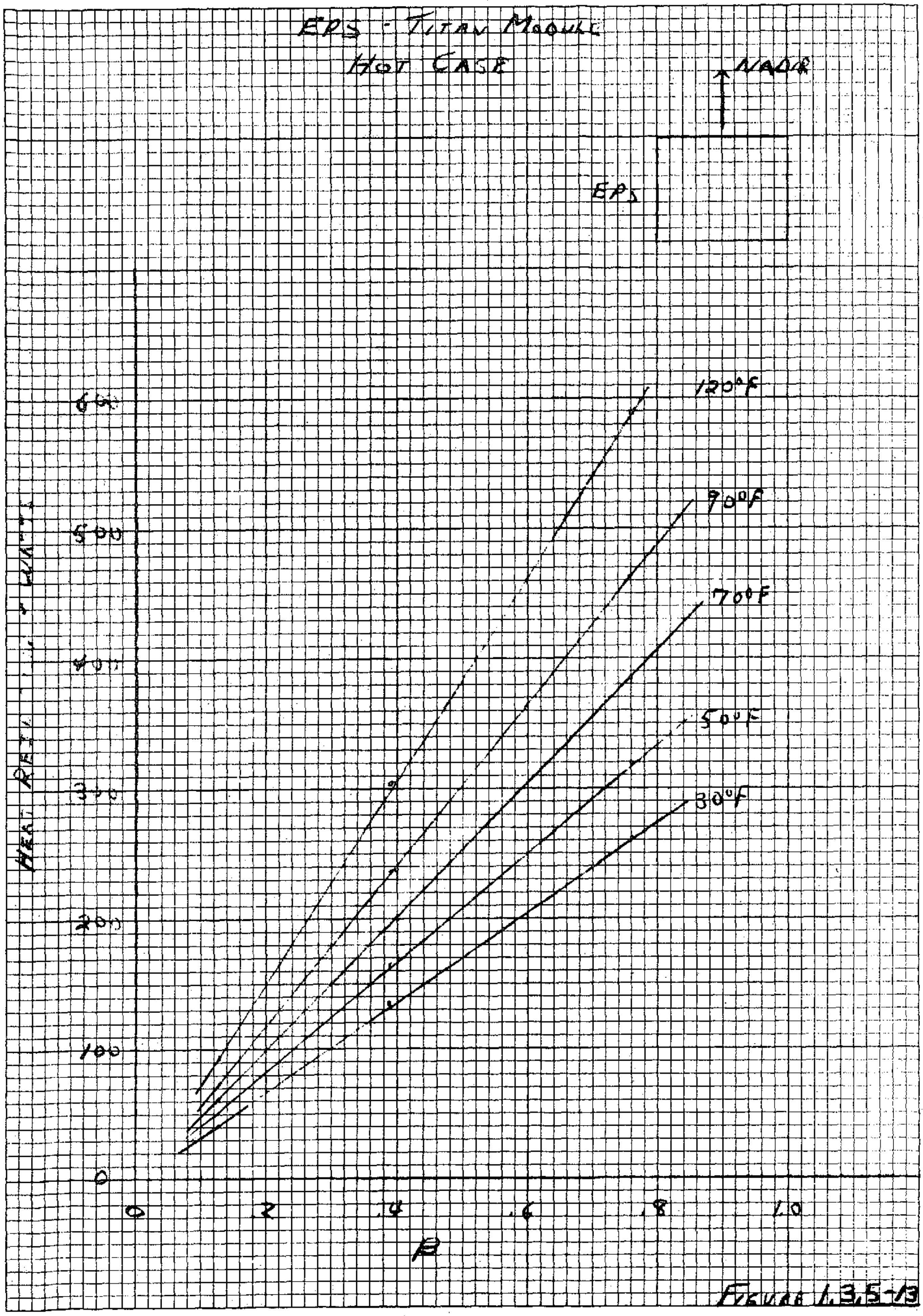


FIGURE 1.3.5-13

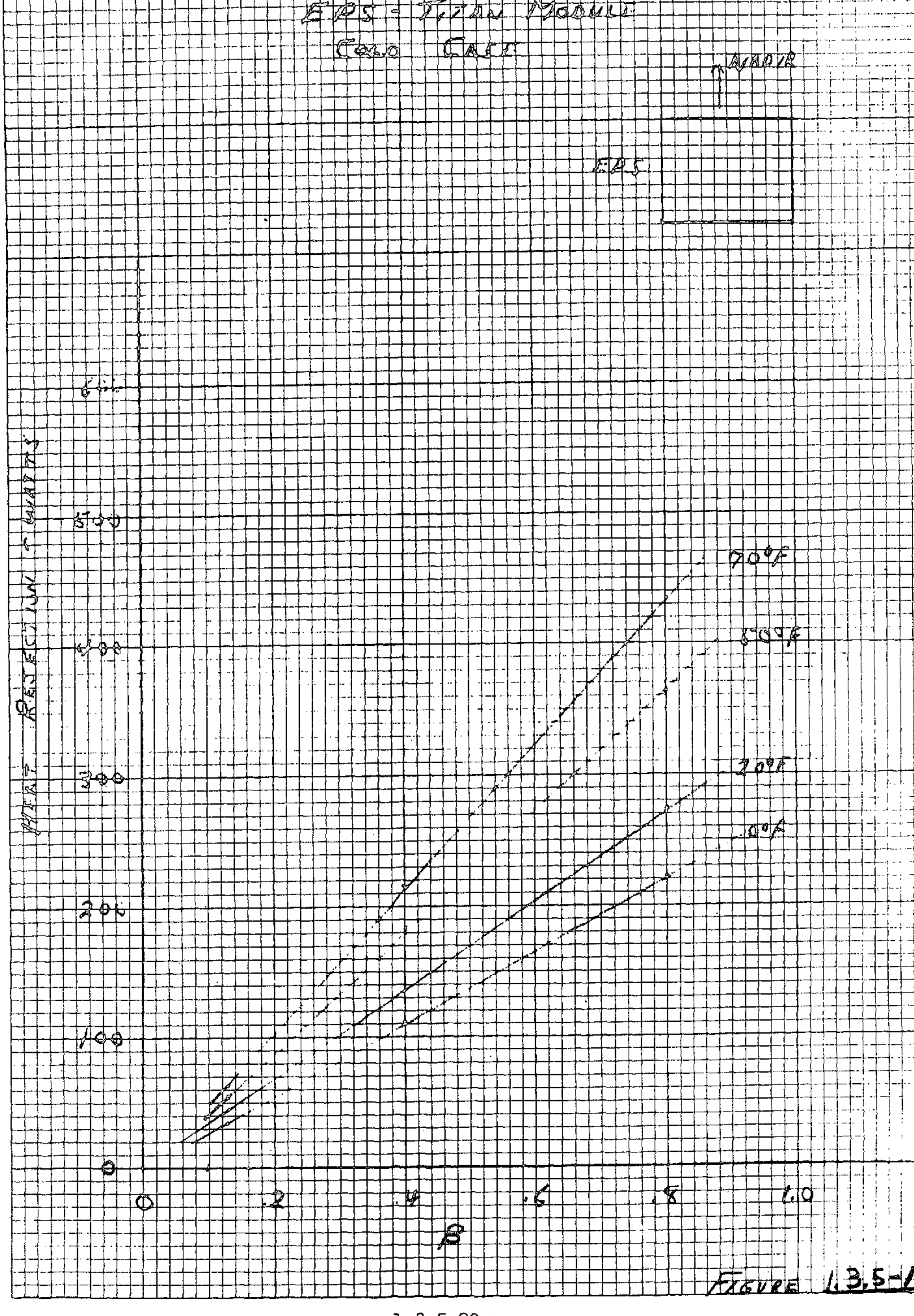


FIGURE 1.3.5-14

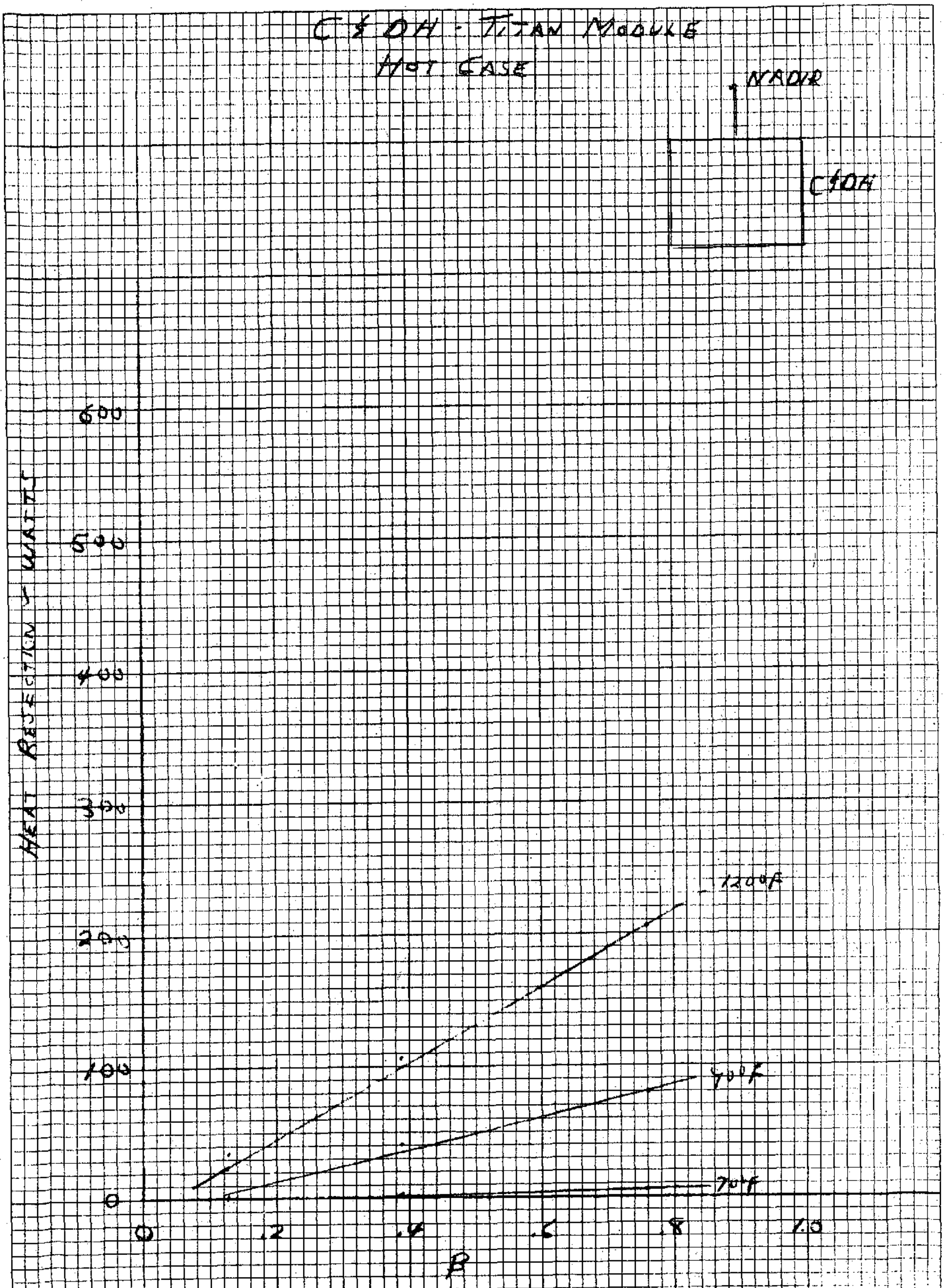


FIGURE 1.3.5-15

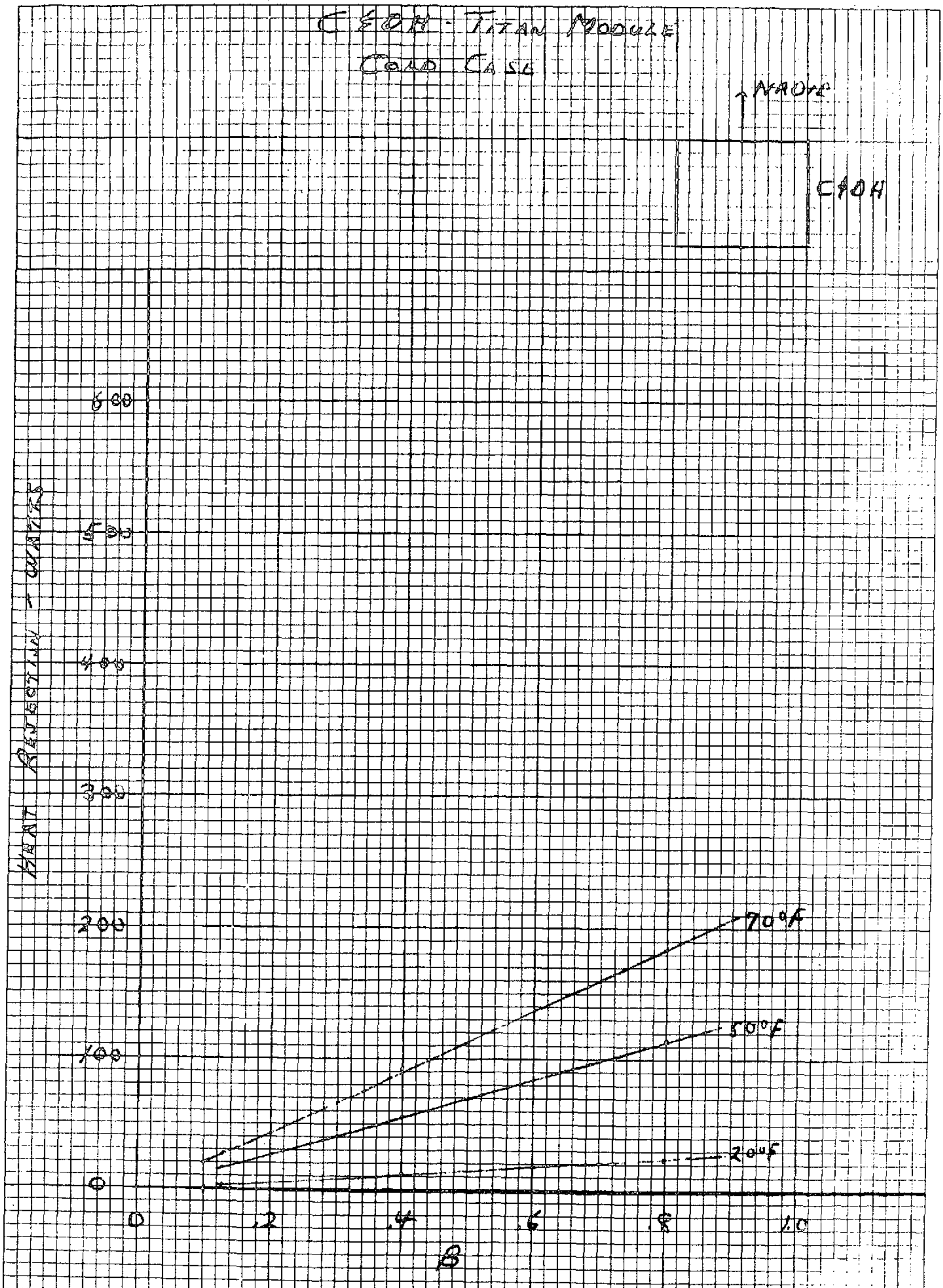


FIGURE 1.3.5-16

1.4.1 WIDEBAND DATA HANDLING AND COMPACTION

1.4.1.1 INTRODUCTION

The function of the S/C wideband data handling and compaction subsystem is to convert, format, multiplex and select multichannel analog video data from multispectral scanning instruments and produce serial digital NRZ data streams at suitable rates for transmission to primary and low cost ground stations via radio links.

Figure 1.4.1-1 depicts the overall data handling subsystem block diagram. The functions shown within the dotted lines are the data handling subsystem functions required to handle the baseline instrument payloads and include appropriate commandable switching functions to apply otuput data to WBVTR/TDRS, QPSK modulator or BPSK modulators. The data handling subsystem will have appropriate interfaces with the instruments, S/C prime power, S/C on-board computer, the WBVTR/TDRS option function and the direct primary ground station or LCGS radio link modulation functions. In addition, an auxiliary low data rate interface is envisioned to handle appropriate low rate S/C telemetry and /or PMMR instrument data to the extent that it can be inserted during available TM overhead format time. A speed buffer function is included to provide for a partial scene data compaction option for either TM or HRPI. In general the output rates from either TM or HRPI data handling units will be constrained to be equal at a value R megabits per second. Similarly the compacted rates from either instrument will be constrained to some convenient rate R/α megabits per second. Due to the high rate (R) and the probable physical separation of units from the QPSK modulator, in-phase (I) and quadrature (Q), retiming functions are included to properly condition the NRZ signals for QPSK modulation. The synthetic aperture radar (SAR) signals will also be constrained in rate to equal value R to provide compatibility and commonality of modulation equipment. Subsequent paragraphs will treat the exact alternative approaches to dividing and modularizing the circuitry.

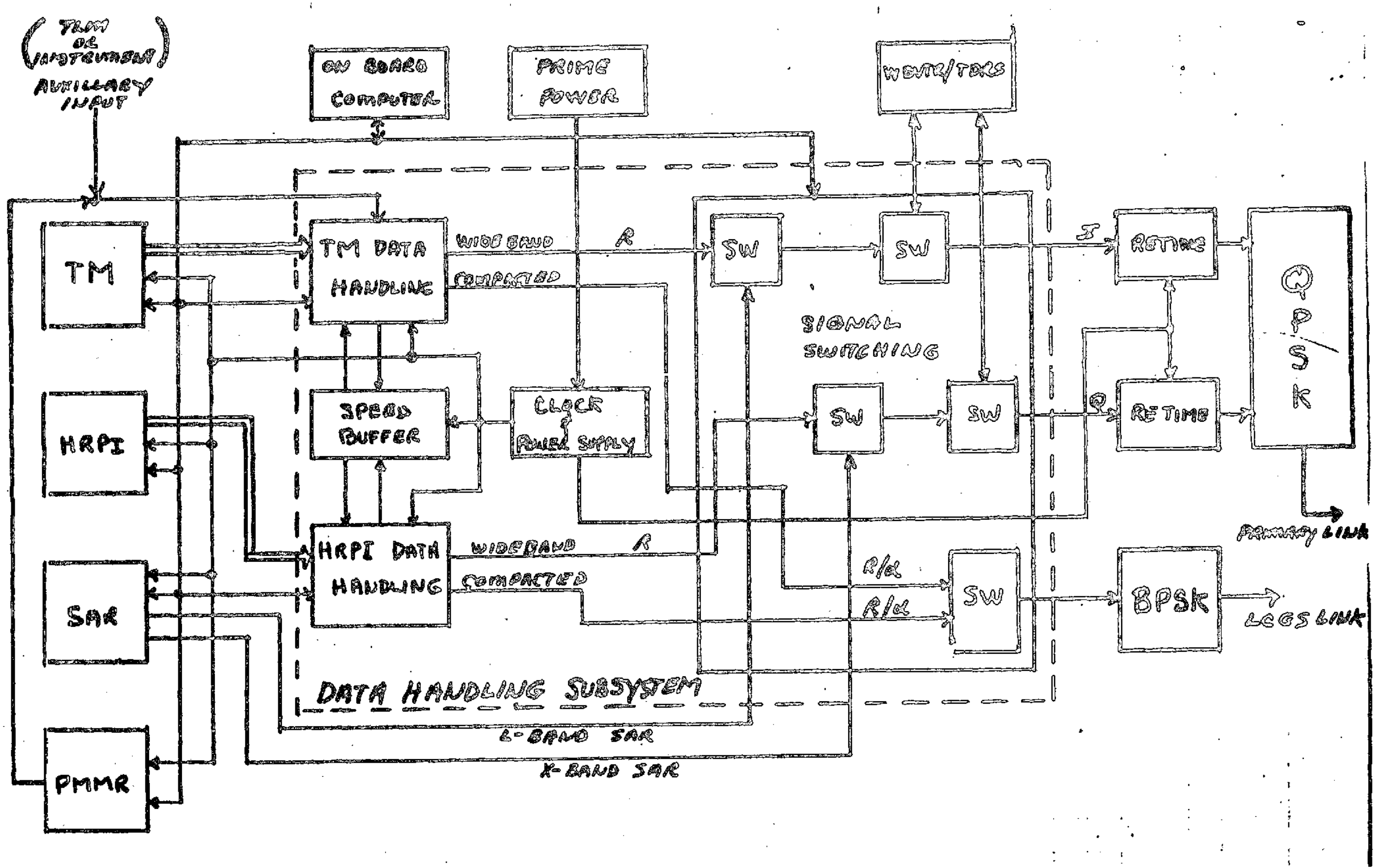


Fig. 1.4.1-1 Data Handling Subsystem Functional Block Diagram

1.4 1.2 ALTERNATIVE SUBSYSTEM CONFIGURATIONS

There are several factors contributing to the configuration determination of the data handling and data compaction subsystem. These are listed as follows:

1. Instrument manufacturer's desire for a digital interface.
2. Magnitude of the digital rate at the instrument interface (if digital).
3. Size of the electronics package that can be placed inside or in external contact with the instrument.
4. Consideration for multiple instrument data handling function requiring modular flexibility.
5. Consideration for multiple instrument data compaction where only one compacted instrument output can be selected and sent at one time.
6. Presence or absence of high capacity speed buffering in the data compaction functions.

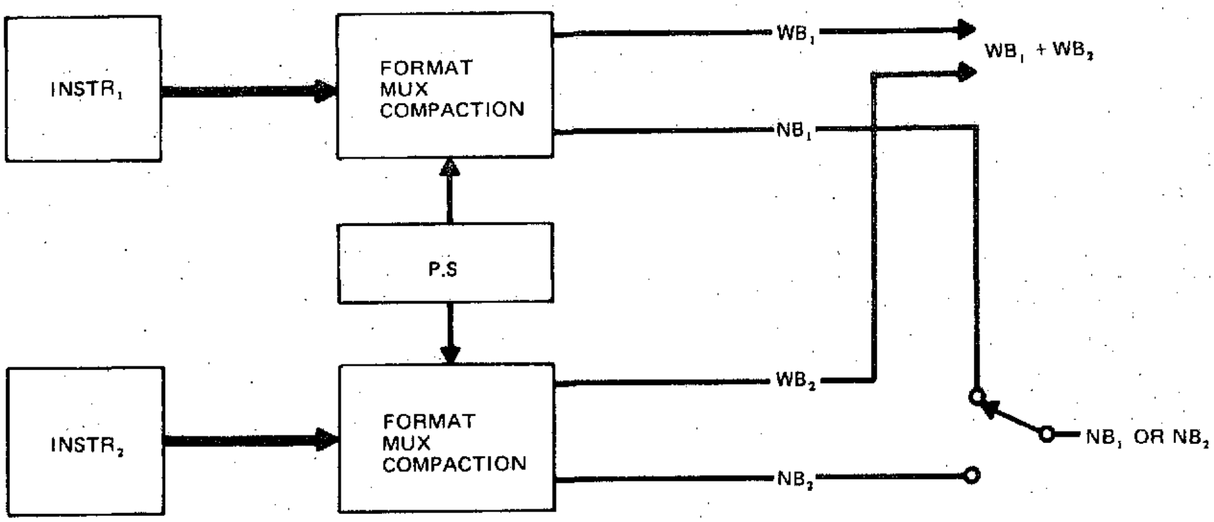
Figure 2-2 portrays the single instrument data handling and compaction function alternative that could be considered for EOS. Fig. 1.4.1-2(a) shows an analog interface (multiple bands and multiple detectors per band) feeding a data multiplexer, formater and A/D converter. Analog cabling run lengths of several feet are assumed for this case and the number of analog lines could run as high as 100 per instrument. The Radiation Inc. MOMS point design assumed such a configuration with analog inputs. In discussions with Hughes Aircraft instrument people it became apparent that they would prefer to have a digital interface with the instrument due to the difficulty in guaranteeing performance at the end of long analog lines. Their concept of this would be to provide a single digital serial bit stream per color band for either the TM or HRPI at a moderate data rate ($\approx$ 12 Mbps for TM and $\approx$26 Mb forHRPI). The electronics for multiplexing and A/D conversion would either be supplied by them or someone else. The key point being that they (Hughes) would be responsible for the instrument output performance to the digital per band interface level. This low rate digital instrument interface is depicted in Figure 2-2(b). It is assumed that the electronics associated with the instrument is in very close proximity to the instrument and is tested and delivered with the instrument.

A logical extension of Fig. 1.4.1-2(b) concept is to include the entire data handling function in the electronics package to be delivered with the instrument as shown in Fig. 1.4.1-2(c). This approach however requires a high digital data rate interface to be reckoned with by the instrument manufacturer ($\approx$ 100 Mbps) and is less attractive from this viewpoint. However only a single data line (two with clock) would be required.

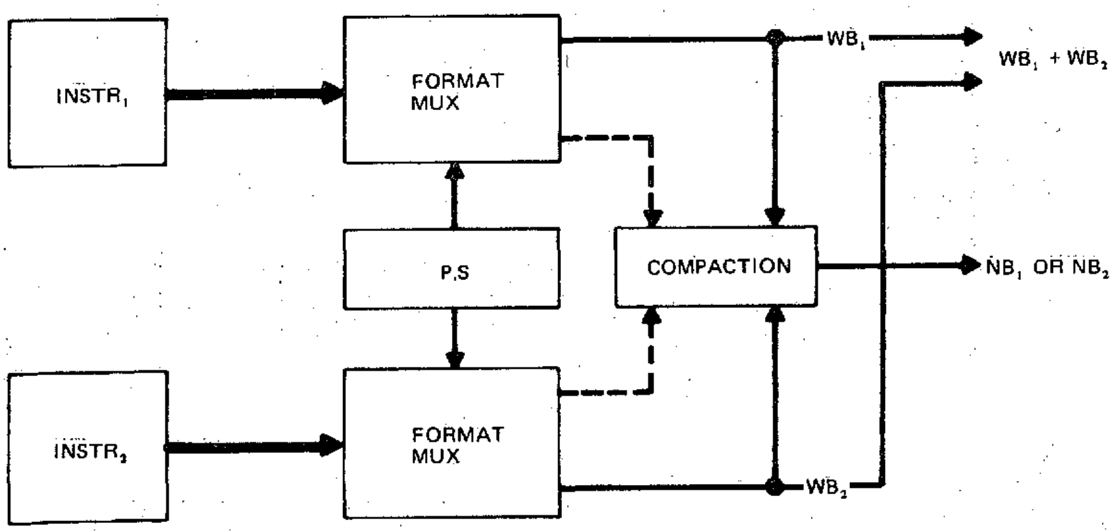
All the above configurations Fig. 1. .1-2(a), (b) and (c) have assumed that the data compaction function is separate from the handling function. There are options here as to where to sample the wideband sensor data for compaction to the lower narrow band rate ($\approx$ 20 Mbps) which depends on the data compaction options implemented and whether a partial scene option is included, thus requiring a data buffer. Simple options requiring no data buffer could be treated differently than those that included a partial scene option.

A final concept that could also be considered follows directly from this discussion and Fig. 1.4.1-2(c). That is, to combine the entire data handling and compaction functions in one unit in close proximity to the instrument.

Fig. 1.4.1-3 shows the considerations of multiple instrument data handling and data compaction. At the outset of the EOS study work it was supposed that multiple instruments meant more than two. However, it soon became apparent that only two instruments (TM and HRPI) existed that require high rate data handling and data compaction. The desire for modularity andflexibility have indicated separate formating and multiplexing modules for the TM and HRPI. The data compaction function is another matter. If no partial scene compaction option is offered, no buffer is required and the functions for band selection or detector averaging can best be combined with the respective instrument data formating and multiplexing functions. If partial scene options are offered for each instrument, the large buffer required makes a separate shared compaction function module an important candidate consideration. Another possibility is to just provide a common shared buffer module with the remaining



(a) WITHOUT COMPACTION BUFFERING



(b) WITH COMPACTION BUFFERING

D-1

Fig. 1.4.1-3 Multiple Instrument Alternatives compaction circuitry located with each instrument formating-multiplexing module. It is not yet clear whether the HRPI compaction function should include a partial scene option. If it does not, any buffer would be strictly associated with the TM instrument with appropriate modification to the argument for a separate compaction buffer circuit module.

Block diagram examples in more detail for the analog and low rate digital instrument interface cases are shown in Fig. 1.4.1-4 and -5 respectively. Fig. 1.4.1-4 is essentially the MOMS approach for the analog digital interface and a separate compaction (with speed buffer) module approach. Note also that the compaction function with speed buffering is shared with the TM or HRPI in in Fig. 1.4.-14. In the original MOMS point design only TM was assumed to be compacted. Fig. 1.4.1-5 depicts the block diagram for a TM wideband data handling and compaction unitsfor the low rate digital interface case. The compaction circuit shown is integral with the formatting and multiplexing functions corresponding to the Fig. 1.4.1-3(a) concept. At this juncture it is not clear whether partial scene compaction is to be employed. Hence, the storage block in the lower right hand corner of Fig. 1.4.1-5 may or may not be a significant hardware item. If it is and both TM and HRPI compaction is involved then a separate compaction circuitry/buffer module may be entertained in accordance with the Fig. 1.4.1-3(b) concept.

Fig. 1.4.1-5 represents the latest thinking in the data handling implementation configuration. The HRPI handling and compaction would appear similar to the TM handling and compaction shown in Fig. 1.4.1-5 with appropriate modification to the number of input color bands and the detailed manner of inserting overhead information (formatting).

1.4.1.3 DATA HANDLING AND COMPACTION ISSUES

A. Instrument Interface

Three alternatives exist for the instrument interface. These are: (1) analog, (2) non-formatted digital plus control lines and transparent digital containing all

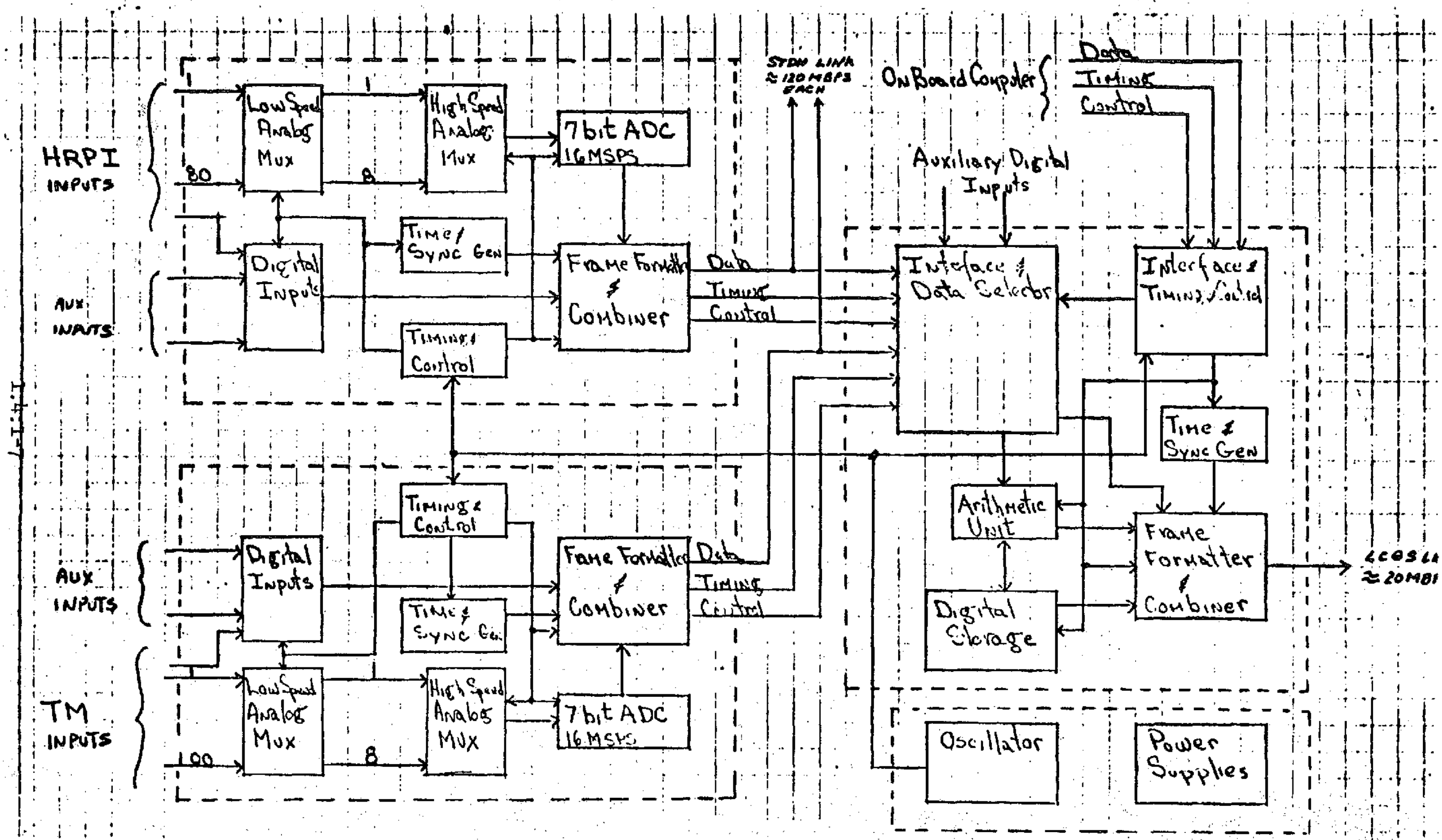


Fig. 1.4.1-4 Wideband Data Handling and Compaction Subsystem with Analog Instrument Interface

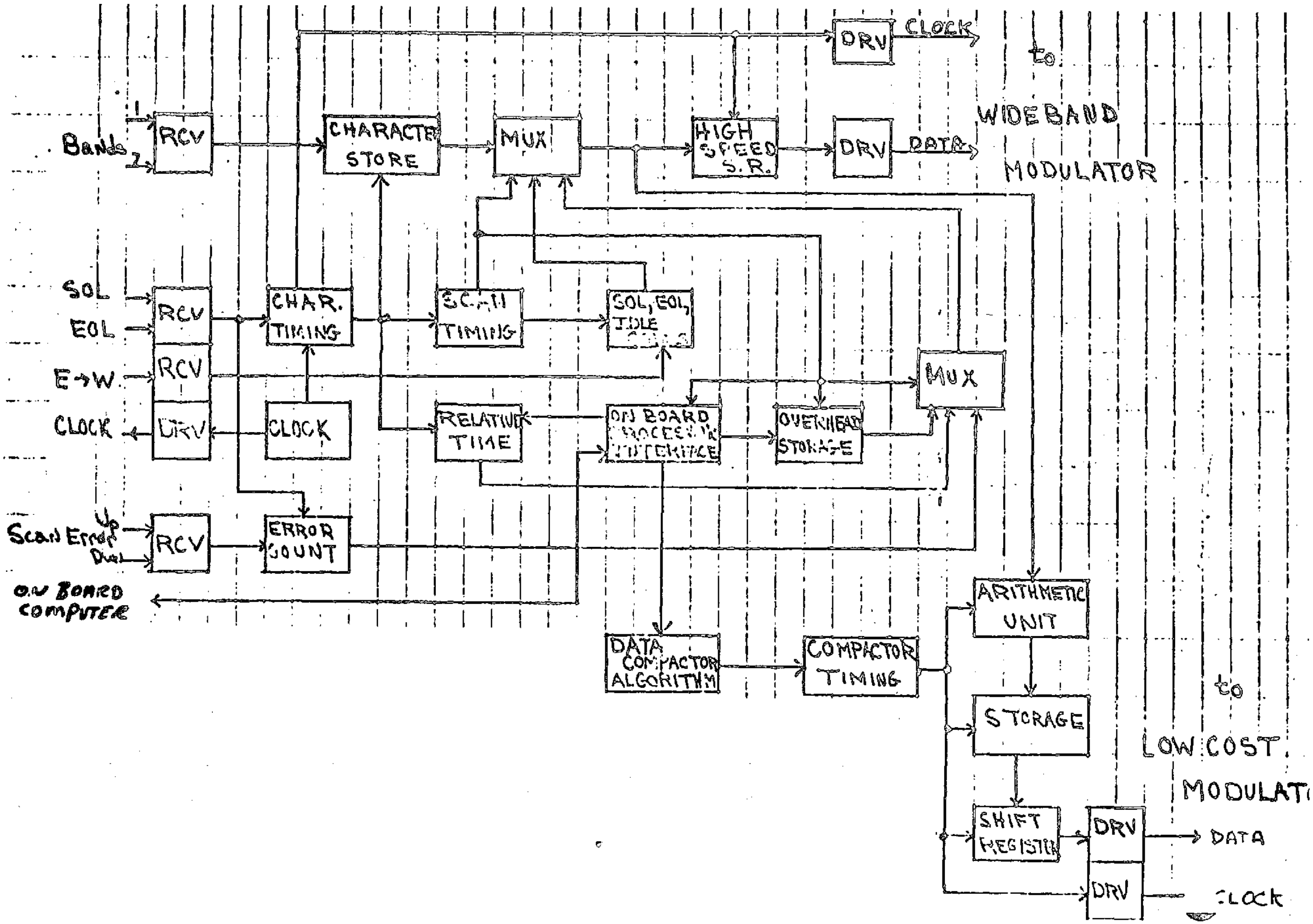


Fig. 1.4.1-5 Wideband Data Handling and Compaction Subsystem with Low Rate Digital Instrument Interface - (TM Case Shown)

required overhead. The analog interface has tentatively been eliminated primarily from the instrument manufacturers point of view. The remaining approaches are both acceptable to comply with a digital interface. Limited time has forced the assumption of the non-formatted digital plus control line approach for this interface thus performing all of the formatting and combining of overhead information in the data handling unit for the respective instrument.

B. Overhead

Included in each line with the instrument sensor data is a small amount of non-sensor data. This non-sensor data and that part of the sensor data not occurring during the active scan interval are referred to as overhead. The latter can be used for optically calibrating the instrument. An electrical calibration of the post sensor electronics can also be obtained during this time.

In both the TM and scanning HRPI, the active scan interval (the interval between the start of line (SOL) and end of line (EOS)), is eighty percent of the scan time. This allows twenty percent for calibration data and the non-sensor overhead. Five to ten percent should be sufficient for the calibration.

The active scan is initiated at the SOL signal or some short known time after it. A code is inserted in the output bit stream in place of the sensor data that identifies the SOL. Additional overhead data may also be inserted after the SOL code, e.g., relative time. At the EOS another code is inserted to identify the end of the active scan. Additional overhead may also be inserted at this time.

Ground processing of the received wide band data starts with the detection of the SOL code. To make this code stand out, an idle code can precede it. This code would provide a good background against which the SOL code can easily be found. The SOL code should provide the maximum distance from the idle code to allow error correction and prevent false acquisition on noise. A simple combination would be a seven bit code and its complement. However, such a short code could cause an LCGS radio link flux density problem. A PN sequence could also be used which would not create the flux density problem, but would be more costly to implement. Start and end of lines should be accurate to at least 0.1 pixel elements. This allows these codes to occur on character boundries (7 bits) within the output data stream. It appears that a seven bit code is sufficient for SOL, but EOL should be 14 or 21 bits since it does not follow an idle code.

In order to process the data on the ground, parameters such as pitch, roll and yaw and their time derivatives are required. This information is available in the on-board processor (OBP) and would be supplied to the data handling equipment. Its inclusion is probably best accomplished after the EOL code and before the calibration data.

Another requirement for ground processing is the elimination of scan non-linearities. It is assumed that a scan error signal can be derived from the instrument. Such a signal could be a delta modulation signal that can be accumulated in the data handling equipment or sent as it is received. Overhead slots exist in the format of the TM during the active scan time due to the reduced resolution of band 7. The accumulated error can easily be sent during these times. In the HRPI no overhead exists during the active scan (unless the total wideband rate exceeds four times the single band rate) and either a data bit has to be preempted now and then or the scan error must be transmitted after EOL or calibration. The latter approach requires a large buffer to store the error signal. In the former case preempting a single bit every second pixel would suffice, and require a buffer of less than four thousand bits. Another implementation of the former case could be to preempt the least significant bit of a particular detector substituting the scan error signal. The detector A/D accuracy is reduced but this may not be a problem especially if seven bit quantization is used.

C. Rates

1.0 Wideband Rate

The data rate from the instrument is determined by a number of parameters. For a square pixel, the pixel dimension or resolution has the largest effect. Spacecraft altitude has the least effect, a variation of about 10 percent for an altitude of 500 to 1000 kilometers. The equation below expresses the bit rate as a function of these parameters.

$$R = \frac{Vg \cdot Dg \cdot S \cdot N \cdot K}{(RE)^2 \cdot E}$$

where Vg = ground speed of spacecraft = $\dfrac{7.91}{\left(1 + \dfrac{h}{6378}\right)}$

Dg = ground distance covered in a scan

S = number of bits per sample

RE = dimension of resolution element

E = instrument efficiency

N = number of bands per instrument

K = number of samples taken for each resolution element in scanning direction.

h = S/C orbit altitude

In the current design, the parameter K seems to be the one whose value is least certain. With an assumed value of $K = 1$ and

$h = 680$ Km

$Dg = 185$ Km

$S = 7$ bits/sample

$N = 7$

$E = 0.8$

$RE = 30$m

the rate, R, for the TM is 85.75 mbps. With the TM and HRPI at the same wideband rate and

K = 1 sample/resolution element

S = 7 bits/sample

N = 4

E = .8

RE = 10m

the ground distance covered by a HRPI scan is about 36 Km. This is about 25 percent less than the point design of the HRPI, a result due mainly to the switch from a high efficiency (100 percent) push broom HRPI to the lower efficiency scanning HRPI.

2.0 Compacted Data Rate

The compacted data rate should be in the order of 20 megabits per second. At the wideband rate of 86 mbps a division by four produces a rate of 21.5 mbps. This is easily produced by counting down the wideband lock and as indicated in Section 2.3.4, provides for some very good compaction schemes. The actual rate used for compacted data should be examined from both the LCGS radio link and S/C data handling viewpoints. Increased cost for a two dB increase in power, e.g., to handle 30 mbps rather than 20 mbps, may be more than offset by decreased complexity in the data handling equipment.

D. Compaction

Data compaction schemes with a digital instrument interface require only digital processing techniques. Three types of compaction are considered for both the HRPI and the TM; band selection, resolution reduction and partial coverage. The specifics of these are obviously impacted by the rate used and the amount of buffering available.

Resolution reduction from 10 meters to 20 meters in the HRPI and from 30 meters to 60 meters in the TM decreases the data rate to one quarter of the wideband value. This is the minimum reduction and one quarter rate (20 to 30 mbps) can probably be supported by a low cost ground station. With this compaction scheme, very small buffer storage is required, resulting in about one half the number of bits produced each sample time. These factors favor a compacted rate of one quarter the wideband rate.

C-4

Band selection for the HRPI is rather limited since there are but four bands. The only feasible selection is a single band, and this would require a rate one quarter of the wideband. This is another factor supporting the one quarter rate recommendation. For the TM a combination band selection and resolution reduction would provide a single band at 30 meters resolution and three bands at 60 meter resolution at the same rate. As an alternative, two bands at 30 meters resolution could be provided at two sevenths of the wideband rate. This rate, however, is not compatible with the HRPI unless the data is buffered and fill or overhead inserted into the bit stream.

Band selection and resolution reduction described above at one quarter the wide band rate are attractive since they do not require a large buffer memory to smooth the difference between the input and output rates. The compaction hardware could be included in the same physical package with the wideband, reducing interfaces and power dissipation.

The third alternative, partial coverage requires a buffer memory. Its size is dependent upon the number of bands involved, the compacted rate and the amount of coverage. With a compacted rate of 21.44 mbps, Fig. 1.4.1-6 indicates the minimum buffer size as a function of swath width covered for TM and HRPI instruments. Maximum coverage is limited by either swath width in the case of two bands of the TM or amount of data for N = 3 through 7 of the TM and N = 2 through 4 for the HRPI. It is interesting to note that a HRPI reduced resolution supplies a type of TM partial coverage. A swath width of 19 nautical miles or less involving the first four bands of the TM (same as the HRPI bands) is more economically produced by the HRPI than the TM, since no buffer is required for reduced resolution. In order to cover the same swath width for four bands with the TM, a 300 kilobit memory is required and the resolution is degraded from 20 meters to 30 meters.

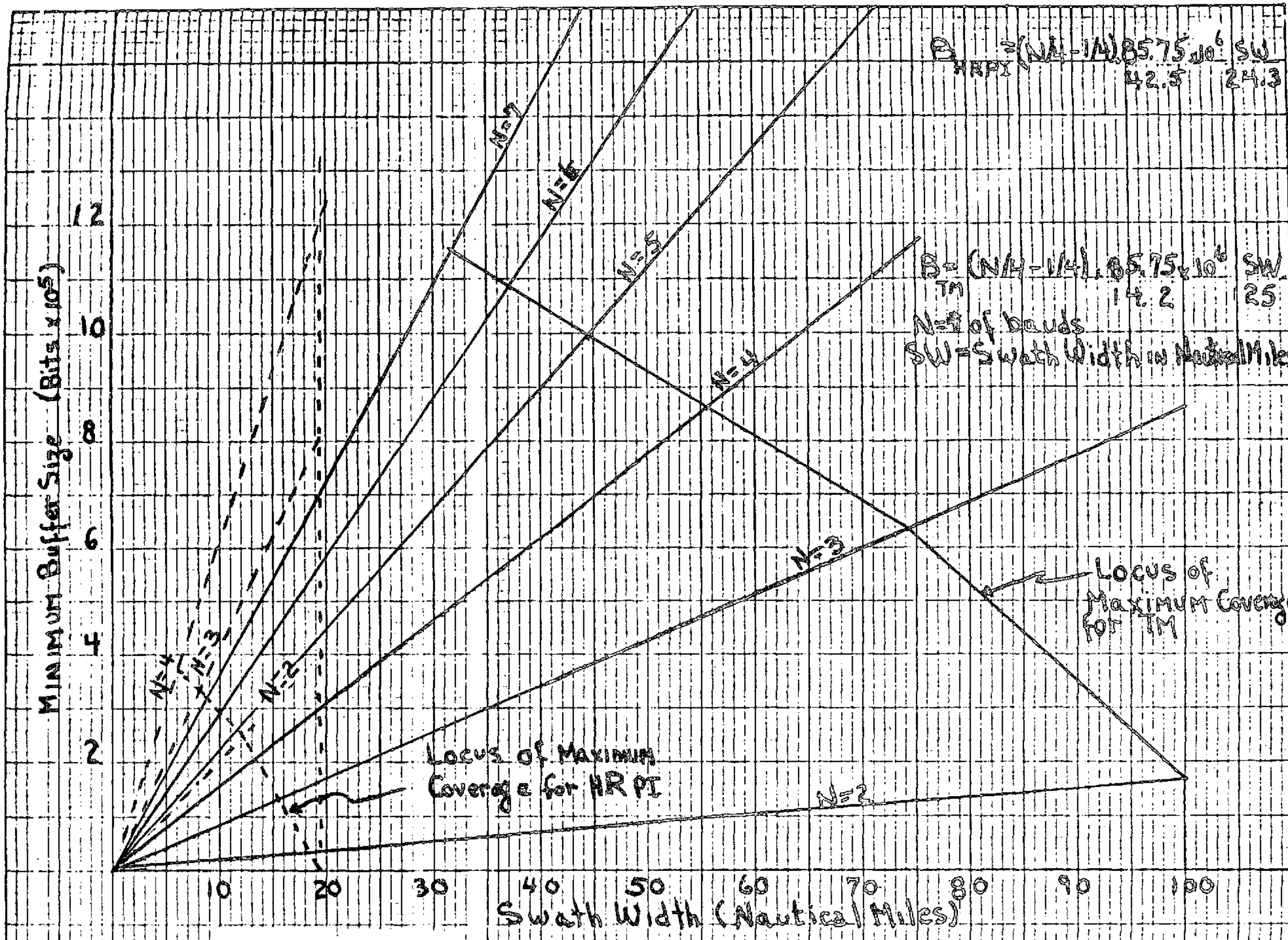


FIGURE 1.4.1-6 PARTIAL COVERAGE BUFFER SIZE FOR TM AND HRPI

1.4.1.4 SELECTED APPROACH DATA HANDLING SYBSYSTEM PRELIMINARY SPECIFICATION

1.4.1.4.1 Introduction

This section is a preliminary specification for a Data Handling Subsystem for the Earth Observatory Satellite (EOS). The EOS mission for the specification consists of two high data rate scanning instruments whose outputs are formatted into two identical rate digital streams for transmission to the ground. These instruments are called the Thematic Mapper (TM) and High Resolution Pointable Imager (HRPI). Two additional lower rate digital data streams,one for each of the instruments, are also required. The data rate is about one quarter that of the high rate (wideband) data and is referred to as compacted data.

The specification is broken down into four areas; interface, functions, power, and clock. Each of these is described in the succeeding paragraphs.

1.4.1.4.2 Interfaces to Data Handling Units

A. General

Three interfaces are defined, as shown in Fig. 1.4.1-7 between the instrument data handling units and instruments, on-board processor and modulators. Data and framing information are supplied via the interface with the instruments. Overhead information required to process the data on the ground and to select the compaction algorithm is received from the spacecraft on-board processor. The third interface is the wideband and compacted data stream to the modulators.

1.0 Instrument Interface

1.01 TM Instrument - TM Data Handling Unit

a. Signals

There are thirteen digital input signals from the TM: seven data, five control and one clock as shown in Fig. 1.4.1-8. The seven data lines contain data as shown in Fig. 1.4.1-9. Each data line, with the

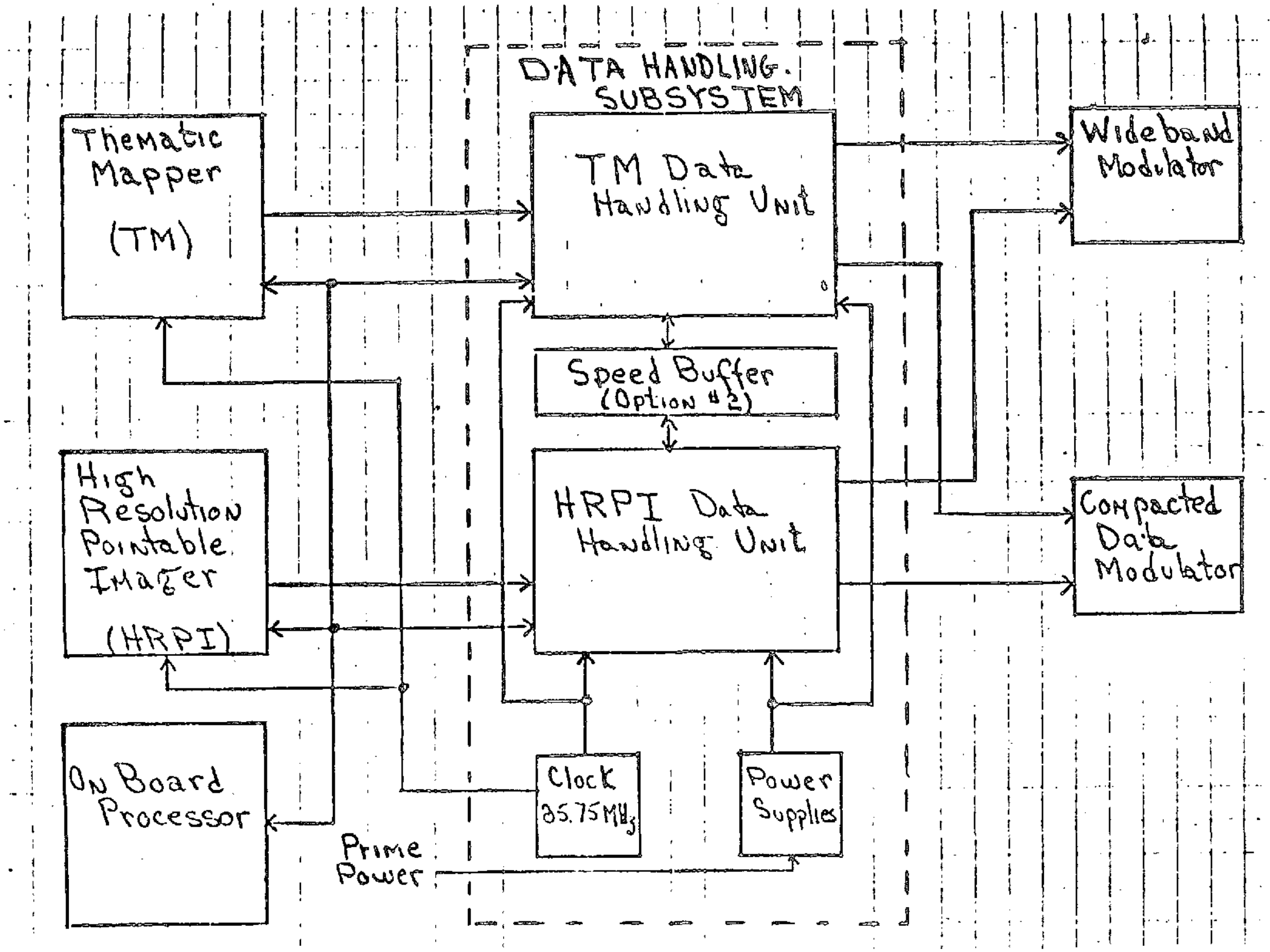


Fig. 1.4.1-7 Data Handling Subsystem Interface

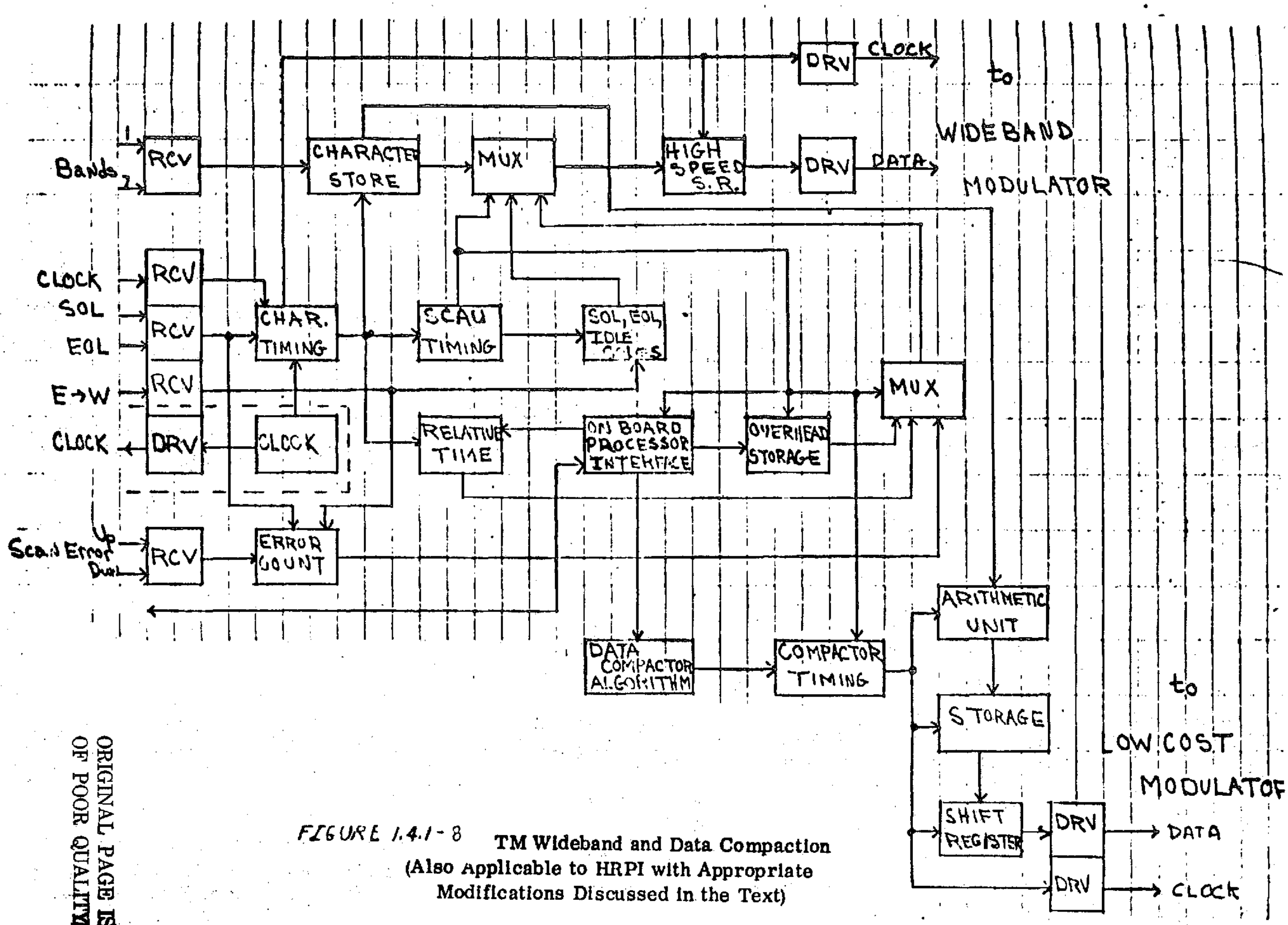


FIGURE 1.4.1-8 TM Wideband and Data Compaction (Also Applicable to HRPI with Appropriate Modifications Discussed in the Text)

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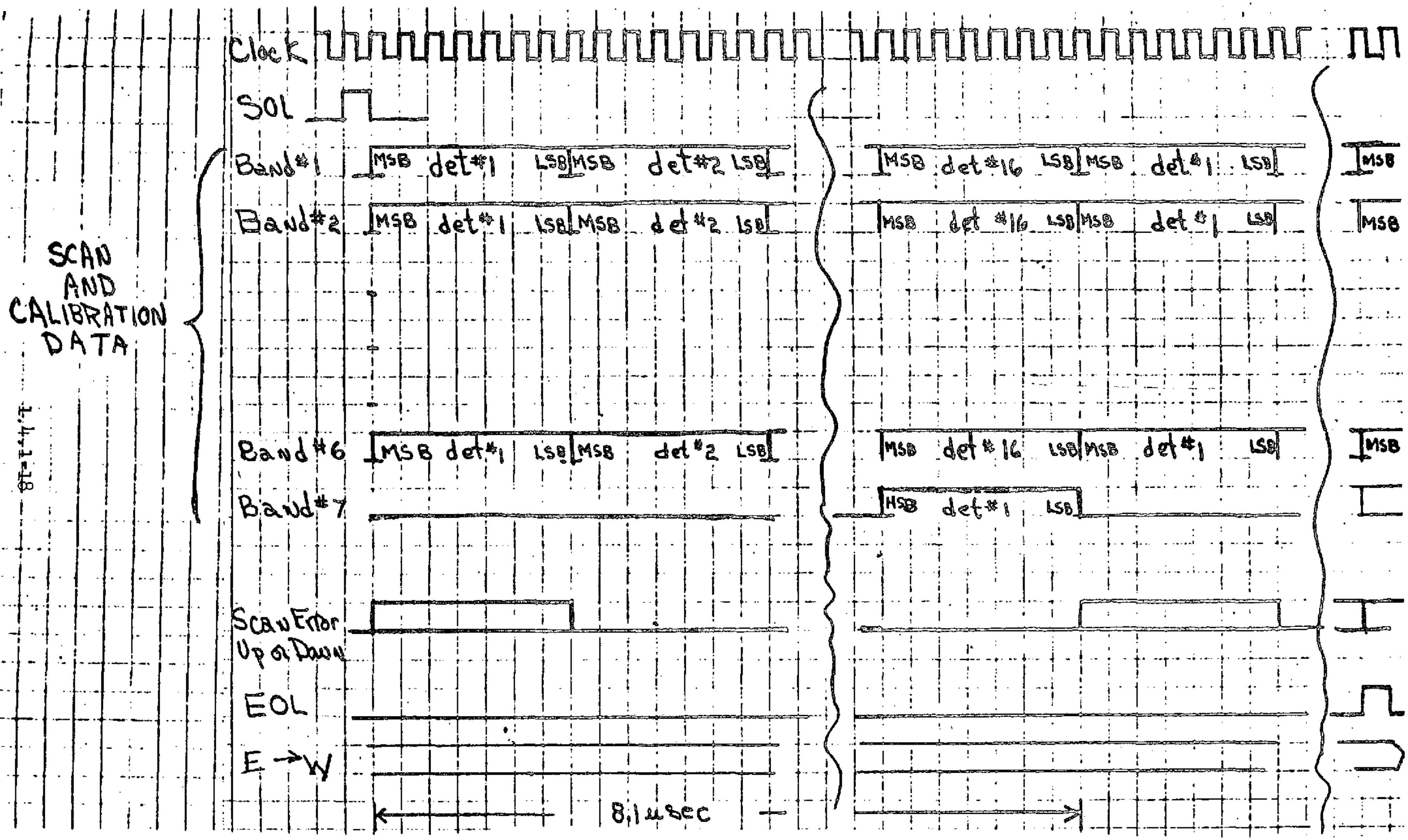


FIGURE 1.4.1-9 TM Interface

exception of the seventh, contains a seven bit sample from sixteen detectors multiplexed together. The seventh line contains a seven bit sample only during the time detector sixteen data is present on the other six data lines. No valid data exists in band 7 during the other 15/16ths of the time.

The five control signals are start of line (SOL), end of line (EOL), east to west scan indicator (E W), and two scan error signals: positive increment (up) and negative increment (down). SOL indicates when the active portion of a scan is started. EOL indicates when the active portion of the scan is completed. E W indicates the direction in which the instrument is scanning. The two scan error signals are incremental change signals that are required to remove scan non linearities in the ground processing of the data. A zero error exists on the east to west scan at the SOL time. The error at any other time in a two scan cycle can be determined by accumulating the incremental error signals.

A clock signal is transmitted from the instrument to define the transition times of the other twelve signals. It may be used to strobe the data and control lines.

b. Rate

The bit rate on each of the seven data lines is 12.25 Mbps. (The average rate on line No. 7 is 1/16th the rates of lines No. 1-No. 6.) The clock signal is a square wave at this rate. SOL and EOL are a single bit interval in duration (82 nanoseconds) occurring at the scan rate, about 14 times a second. E W is a square wave signal occurring at one half the scan rate. Scan error signals are seven bit intervals in duration (570 nanoseconds) and can occur at a maximum rate of one every 8.1 microseconds.

c. Synchronization

1. All signal transitions on the seven data lines and five control lines shall occur on the negative to positive transition of the clock signal within ± 2 nanoseconds as measured at the 50% level of the signal amplitudes.

2. SOL one bit interval in duration, shall occur one bit interval before the most significant bit (MSB) of detector one in bands one through six.

3. EOL, one bit interval in duration, shall occur at the same time as the (MSB) of any detector in bands one through six.

4. Scan error up or down, seven bit intervals in duration, shall occur at the same time as the seven bit sample from detector one on bands one through six.

5. E W, a square wave at one half the scan frequency, shall change state at the bit interval succeeding the EOL signal.

6. Band No. 7 data output occurs as seven bit bursts when detector sixteen's output appears on band one through six.

d. Waveshape

1. Clock - nominally 50% duty cycle with the positive to negative transition within 41 ±2 nanoseconds of the negative to positive transition. Rise and fall times shall not exceed 10 nanoseconds.

2. Data - rise and fall times shall not exceed 10 nanoseconds.

e. Levels - To Be Determined.

1.02 HRPI Instrument - HRPI Data Handling Unit

a. Signals

There are ten digital input signals from the HRPI, four data, five control, and one clock, as shown in Fig. 1.4.1-10. With the exception of the number of data lines, the interface is the same as that of the TM. The four data bands are identical and have the same format as bands one through six in the TM.

b. Rate

1. Data - 21.4375 Mbps per band. This rate is exactly 1.75 times the rate of the TM data lines.

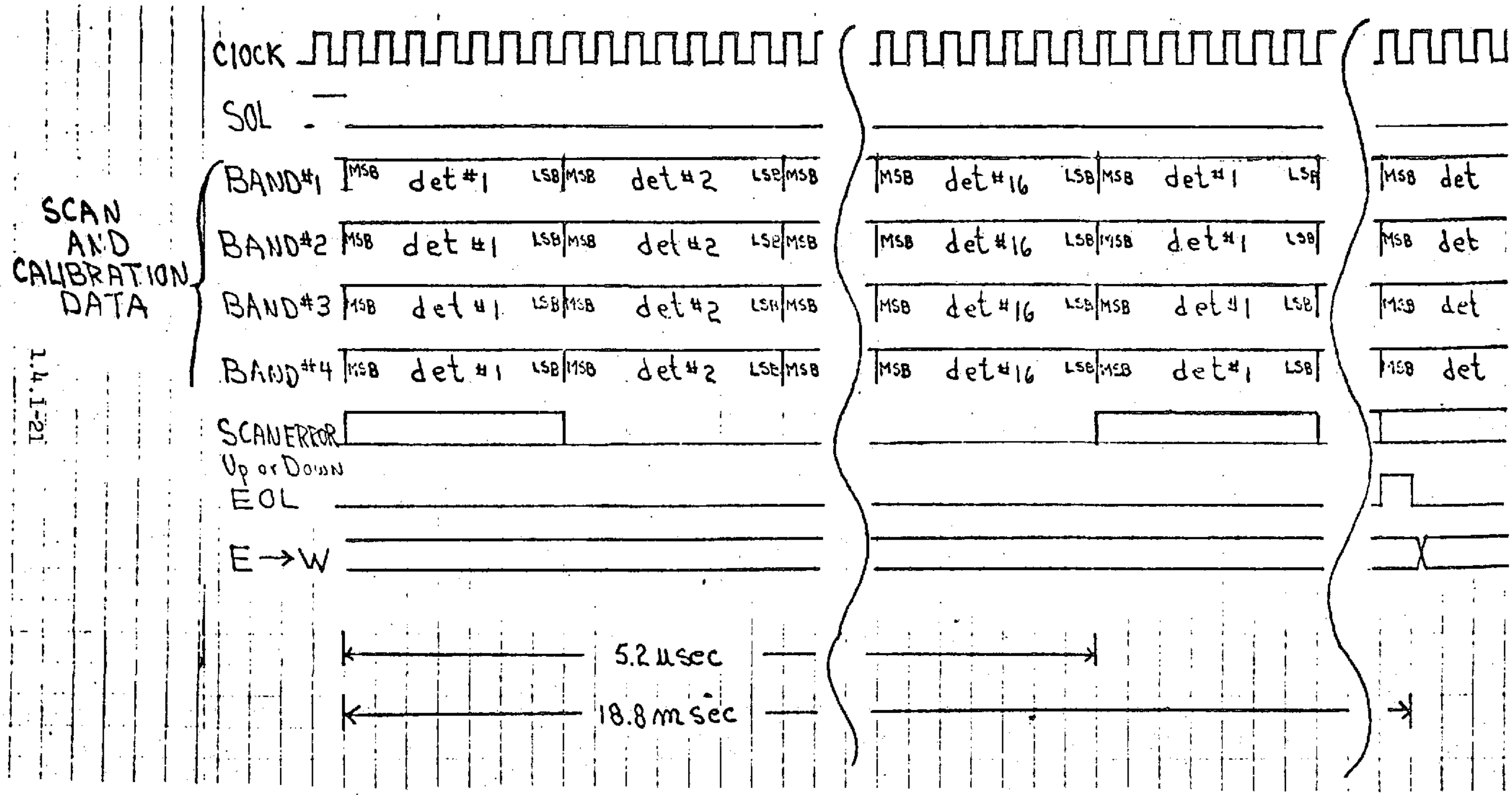


FIGURE 1.4.1-10 HRPI Interface

2. Clock - a square wave at 21.4375 megahertz.

3. SOL and EOL - a single bit interval in duration (46.5 nanoseconds) occurring once per scan interval, 23.5 milliseconds.

4. E W - a square wave occurring at half the scan rate, 21.25 hertz.

5. Scan error up or down - seven bit intervals in duration occurring at a maximum rate of one every 5.2 microseconds.

c. Synchronization

The same relationship applies for the HRPI as for the TM.

d. Waveshape

The same waveshapes apply for the HRPI as for the TM.

e. Levels

The same levels apply for the HRPI as for the TM

1.1 On-Board Processor Interface to Data Handling Subsystem

Communication with the On-Board Processor will be through a remote multiplexer and decoder in the spacecraft command and data handling module. Three input signals are supplied to the Data Handling Subsystem from the decoder; clock (20 kHz) data gate (16 data intervals wide) and serial data. A 16 bit command is received from the onboard processor during the time the data gate is a logic '1'.

Sixteen status outputs are supplied to the processor via the remote multiplexer. These outputs shall include power status, timing status, etc.

The output signal levels from the decoder are:

Logic '1'	+ 12 to + 17 Vdc 4 ma
Logic '0'	0 to + 0.5 V
Source resistance for logic '0'	8,000 ohms maximum

A Siliconix Type SC 299 can be used to receive these signals and convert them to TTL compatible levels.

The input signal levels to the multiplexer are:

Logic '1'	+ 3.5 to + 35.0 Vdc
Logic '0'	- 1.0 to + 1.5 Vdc
Fault Tolerance	- 20 to + 40 Vdc
Source Impedance	5,000 ohms minimum 10,000 ohms maximum

1.2 Modulator Interface

a. Wideband Modulator

The interface to the wideband modulator shall be four signals, two data and two timing from the Data Handling Subsystem. A data and timing signal shall come from each of the TM and HRPI data handling units. The negative to positive transitions of the timing signals measured at the end of 10 feet of appropriate coaxial cable shall be within ±1.5 nanoseconds of each other measured at the 50% point on the signal waveforms. Data transitions shall occur at the negative to positive transition of the timing signal within ±1.5 nanoseconds.

The wideband data rate shall be 85.75 Mbps and be identical for both the HRPI and TM. Rise and fall times of both data and timing waveforms shall not exceed 5 nanoseconds measured between the 10% and 90% of the signal amplitude.

b. Compacted Data Modulator

The interface to the compacted data modulator shall be four signals, two data and two timing from the Data Handling Subsystem. A data and timing signal shall come from each of the TM and HRPI data handling units. The negative to positive transitions of the timing signals measured at the end of 10 feet of appropriate coaxial cable shall be within ±2 nanoseconds of each other at the 50% point on the signal waveforms. Data transitions shall occur at the negative to positive transition of the timing signal within ±2 nanoseconds.

The compacted data rate shall be one quarter that of the wideband rate, or about 21.4 Mbps for both the TM and HRPI data. Rise and fall times of data and timing shall not exceed 10 nanoseconds.

1.4.1.4.3 <u>Functions</u>

A. General

The Data Handling Subsystem has two primary functions, wideband data combining and selective or compacted data combining. In wideband the input data lines (seven for the TM and four for the HRPI) are multiplexed together, framed and combined with a small amount of overhead. These two data streams at 85.7 Mbps (one for HRPI and one for TM) are inputs to a spacecraft QPSK modulator.

Data compaction reduces the wideband data rate to a value that can be handled by a relatively low cost ground station. The data rate selected is exactly one quarter of the wideband rate, or approximately 21.4 Mbps. Data here includes actual data from the instrument plus required overhead and framing.

1.0 Wideband Combining

1.01 TM

The TM produces data during 80% of a scan interval. This data producing interval is defined as the time between the SOL and EOL signals. In addition, about 10% of a scan interval after the EOL contains calibration data on the data lines. The remaining 10% of a scan interval before the next SOL is dead time.

A general block diagram of the TM data handling unit and the instrument inputs are shown in Fig. 1.4.1-8 & 9 respectively. The required output format is shown in Fig. 1.4.1-11. A frame is initiated at

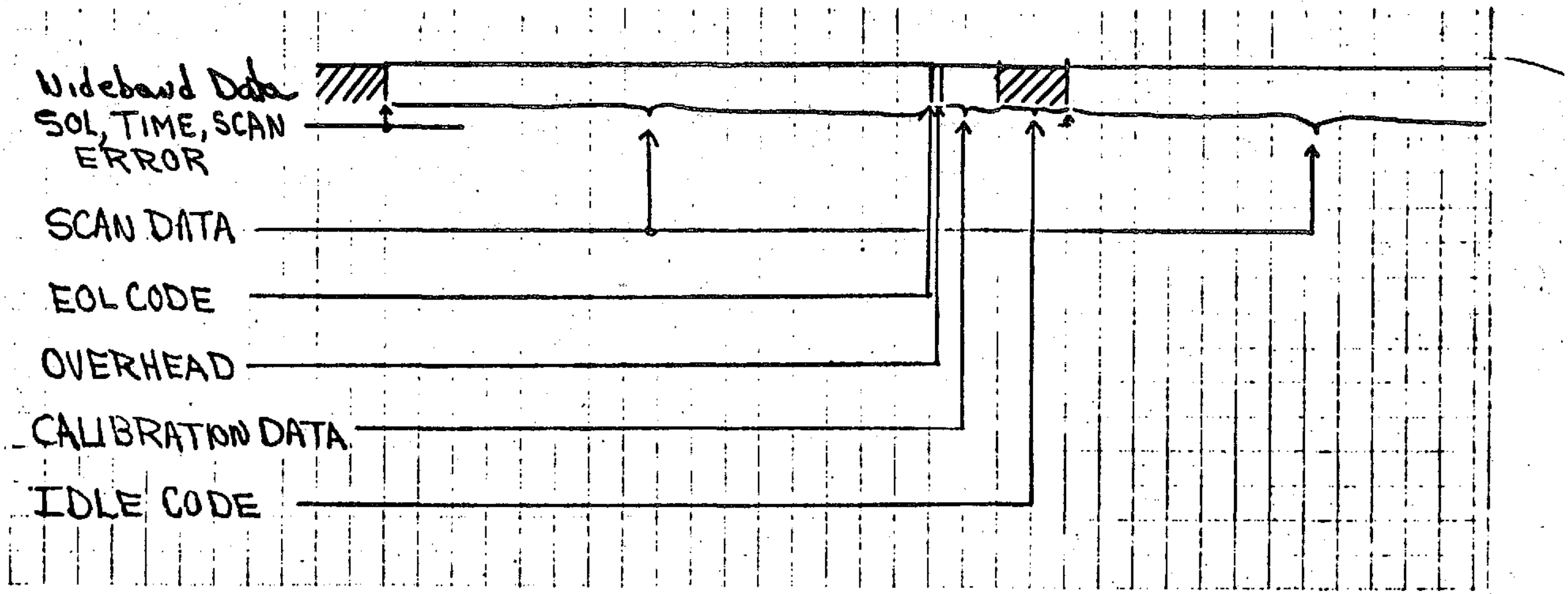


FIGURE 1.4.1-11 Frame Format for Wideband Data

the reception of SOL from the TM. The idle code is removed and a seven bit SOL character code is generated. The next five characters contain the time of arrival of the SOL, from the relative time counter. This counter is preset by the onboard processor and counts the output rate divided by 256. It is a thirty five stage counter. The next character indicates the scan error count at the SOL (0 in E W direction). After these six characters are inserted, inputs from the data lines are multiplexed into the output data stream. The data is combined as it is received, starting at band No. 1 and moving sequentially to band No. 7. Since band No. 7 is mostly idle, it can be used to include the scan error count which is being accumulated. The error function is assumed to be a sinusoidal function ±63 increments peak amplitude. It is to be included in the data stream once every 8.1 microseconds (16 detector outputs on bands 1 through 6).

When the EOL pulse is received from the TM, a two character code indicating EOL and scan direction is inserted into the data stream in place of the incoming data. The next 96 characters are reserved for the overhead gathered during the scan from the onboard processor. After this has been inserted, the data on bands 1 through 7 are again multiplexed into the output data stream for about 7 milliseconds. At the end of this time, a seven bit idle code is continuously applied until the SOL is received and the process repeated.

1.02 HRPI

The wideband functions for the HRPI are similar to the TM with a few exceptions. Duty cycle is 80% and calibration 10% of a scan interval (23.5 milliseconds) as in the TM. No distributed overhead exists however, and scan error must be included in the data stream in some other manner. It could be accomplished by sending a delta modulated signal once every approximately 5.2 microseconds in the LSB of detector No. 16. This bit would start in band No. 1, move sequentially to band No. 4 and then repeat. The calibration time for the HRPI is reduced to 2.4 milliseconds due to the higher scan rate.

1.1 Compacted Data Combining

1.1.1 TM

The format for the compacted data streams shall be similar to that of the wideband. It starts with a seven bit SOL code, followed by five time and one scan error character. Data then follows, the type being selected by the on-board processor. This is followed by the EOL code and overhead. Calibration data may follow depending upon the type of data selected. The idle code follows this until the next SOL.

Two options are proposed, each should be considered for quoting purposes as separate entities. Option one has two modes of operation. The first is an averaging mode, Fig. 1.4.1-12, in which the average value of two consecutive samples of two adjacent detectors in the same band is computed. This single character replaces the four inputs reducing the rate to one quarter of the wide band rate. Band No. 7 is not averaged, its rate is reduced to one quarter by leasing out part of the unused input. Scan error is included in part of the remaining unused portion of band seven. During the calibration time after the overhead has been inserted, the averaging process still applies.

The second mode of option one selects one band at full resolution and three at the reduced resolution indicated above. Scan error is included in band seven if this is one of the selected bands. If not, it shall be included in place of the LSB detector 16 in the highest averaged band.

The second option includes the first option plus the ability to select any number of bands at full resolution and transmit the data gathered during only a part of the scan, Fig. 1.4.1-13. That part of that scan that can be transmitted is dependent upon the number of bands selected. No calibration would be included in this method and a fixed number of idle code characters not to exceed 20 would end the scan format.

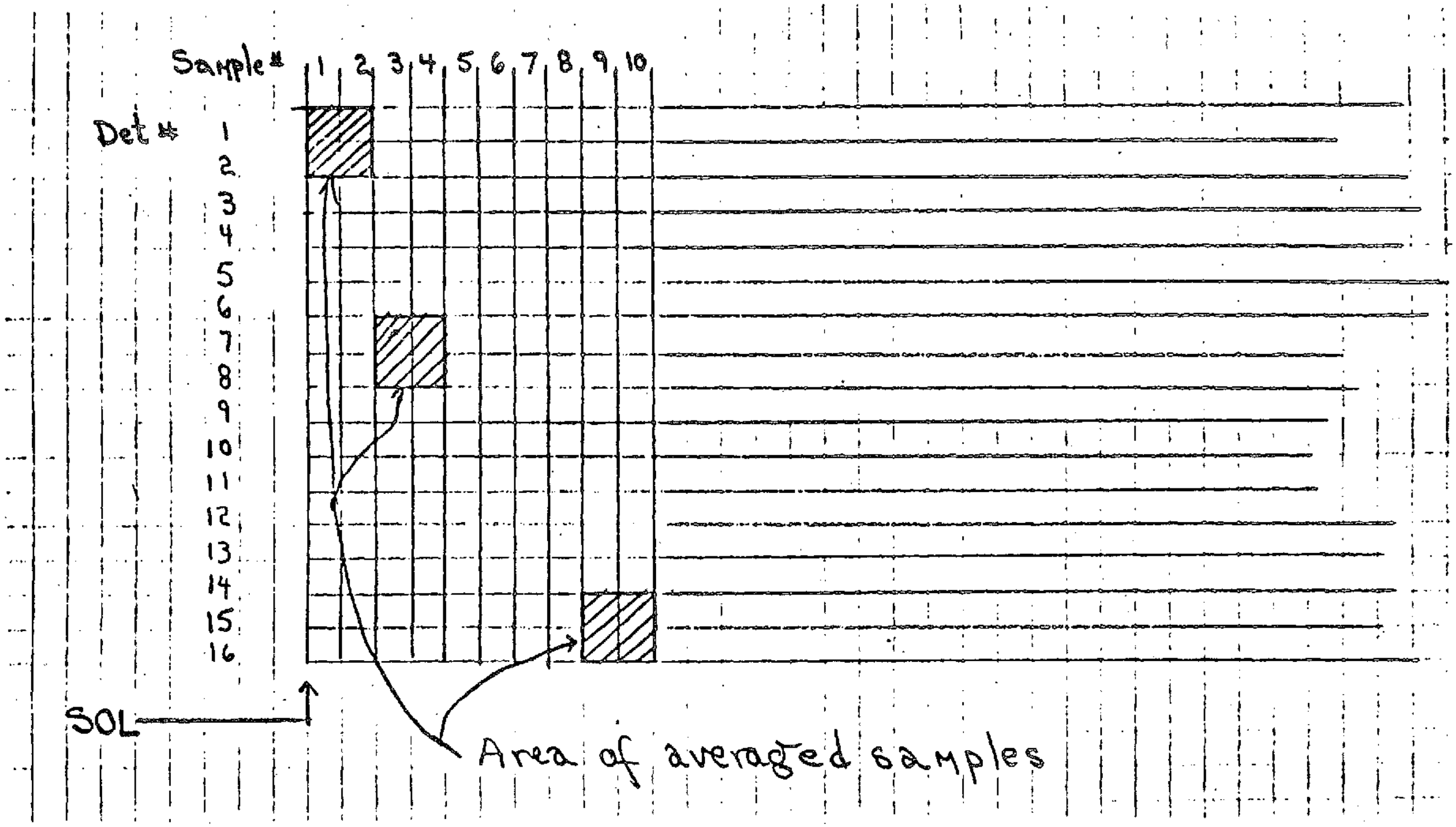


FIGURE 1.4.1-12 Averaging in a Band for Data Compaction

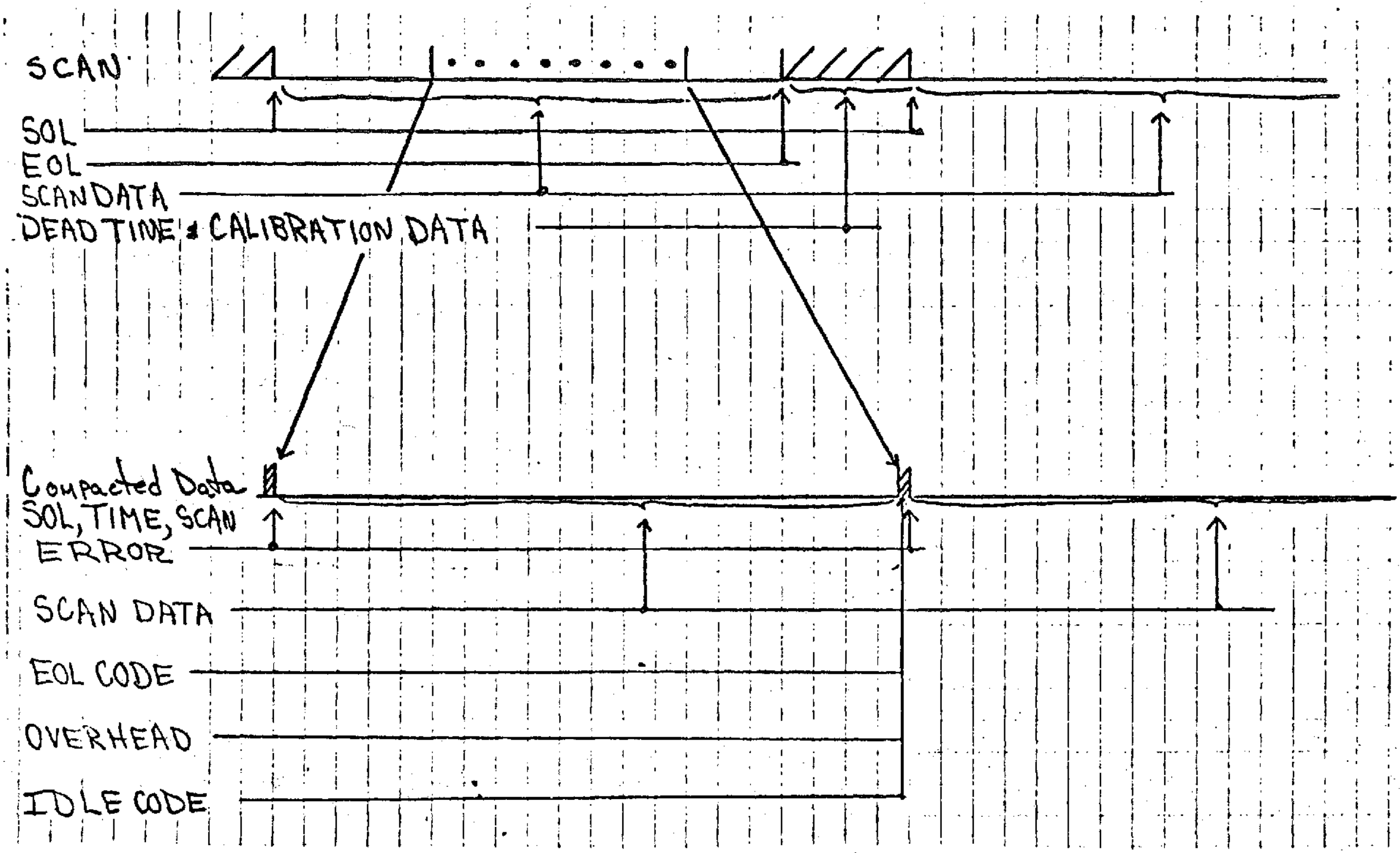


FIGURE 1.4.1-13 **Partial Scan Transmission for Data Compaction**

1.1.2 HRPI

The compacted data format for the HRPI is similar to that for the wideband. Again two separate options are called out for quotation. The first option is two mode. Mode one selects a single band for transmission, reducing the rate to one quarter of the wideband rate. Mode two averages two consecutive points of two consecutive detectors as in the TM to accomplish the 4 to 1 reduction.

The second option includes the first option plus the ability to select any number of bands and transmit the data gathered during a part of a scan at full resolution. The format of the scan data is left to the discretion of the bidder.

1.4.1.4.4 Power Supplies

Power required for the TM and HRPI data handling units shall be supplied by two independent power supplies. Prime power will be 28 volts dc from the spacecraft power buss. All power supplies must be at least 70% efficient.

1.4.1.4.5 Clock

The timing source for the Data Handling Subsystem and the instruments shall be a crystal controlled oscillator at 85.75 MHz. It shall be accurate to 50 PPM and stable to 1 PPM over 30 days. Its estimated MTTF shall exceed that of the remaining components in the acquisition system by at least a factor of ten. Timing signals shall be provided to the TM and HRPI at 85.75 MHz.

D.1.4.2.1 ALTERNATIVE SUBSYSTEM DESCRIPTIONS

Wideband communications is here defined as the direct communication of sensor data, both uncompacted and compacted, from the EOS spacecraft to earth. Two basic links have been identified, distinguished by the magnitude of the rate to be transmitted and the earth terminal resorces desired to receive each transmission. The primary link has been sized at 240 Mbps and is required to be received by STDN earth terminal sites having an antenna as small as 30 foot diameter and a system noise temperature of 166°K. The low cost ground station (LCGS) link has been sized to handle a reduced data rate of 20 Mbps and is to be received by small earth terminals having an antenna diameter in the order of 6 to 12 foot and a system noise temperature in the order of 400° to 600°K, the key issue being low cost.

The basic choice of radio frequency allocation has been made based on the intended operational use of EOS and the available bands for this type of service. Two bands were considered: X-Band (8.025 - 8.400 GHz) and K-Band (21.5 - 22.0 GHz). The inordinately high propagation losses due to atmosphere and rain, and availability of RF components have favored the choice of X-Band.

There are two alternative approaches for establishing the primary and LCGS links. These approaches are depicted in Figure D.1.4.2-1 Alternative 1 employs two narrow beam steerable antennas, 1.6 foot in diameter, each fed by a 4 watt X-Band power source. The philosophy behind this approach is to employ identical S/C components for these links to provide redundancy in case of a failure in either steerable antenna or power amplifier device. It will be seen later on in the discussion that the LCGS link must be power controlled to a lower power than 4 watts to conform to the ITU power flux density limit at the earth's surface sub-spacecraft point. Alternative 2 arises from the desire to eliminate one of the narrow beam steerable antennas, it being replaced with an earth coverage antenna with a shaped beam to optimize the performance

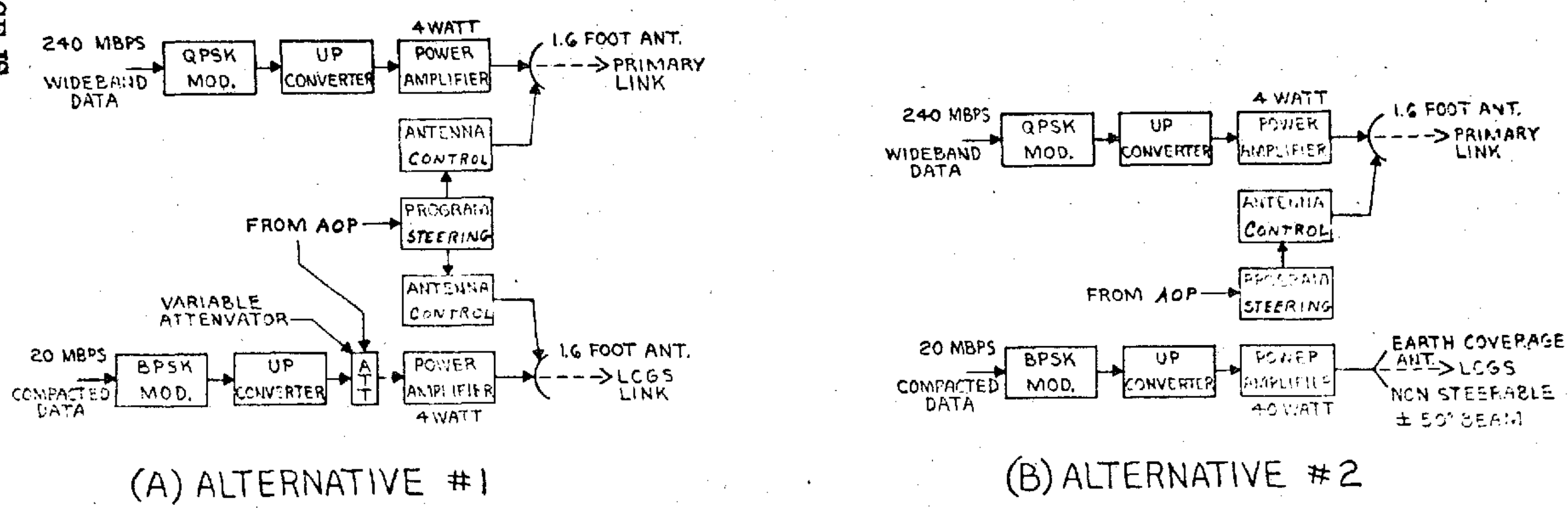


Fig. 1.4.2-1 Basic Wideband Communication Alternative

of the LCGS link at the minimum expected earth station antenna elevation angle. The penalty paid for this approach is a higher power X-Band power source for the LCGS link (40 watts) and a loss of redundancy in critical S/C RF components.

A more detailed discussion of primary and LCGS link tradeoffs and the trade between alternative approaches are given in the subsequent paragraphs.

D.1.4.2-2 ALTERNATIVE SUBSYSTEM PERFORMANCE COMPARISONS

A discussion is in order concerning the primary and LCGS link design choices and alternatives. Several constraints influence the design of these links: the choice of frequency band (X-Band), the power flux density limit at the earth's surface internationally agreed upon for this shared band (8.025 - 8.400 GHz), the available power sources at X-Band, propagation losses, and antenna gains and receiver noise temperatures. Table D.1.4.2-1 tabulates a power budget calculation for both links covering both alternative approaches for the LCGS link assuming the parameters given in Paragraph D.1.4.2-1, Figure D.1.4.2-2 shows the limitations on permissible spacecraft effective radiated power (ERP) as a function of signaling rate based on the ITU power flux density for X-Band. The ERP limit for the 20 mbp LCGS link for a 700 km altitude orbit is 25 dBw and for the 240 Mbps primary link it is 35.8 dBw. The data of Table D.1.4.2-1 as well as variations of this data are plotted in Figure D.1.4.2-3 to enable a discussion of tradeoffs that can be made with respect to both the primary and LCGS links.

Primary Link Discussion

The primary link must transmit the 240 Mbps wideband data to an earth terminal having a maximum antenna size of 30 foot diameter to be compatible with all STDN locations. At this point it is possible that either one of three STDN sites will share the primary link reception or a single new site probably located at Sioux Falls, South Dakota will be constructed for this purpose. If STDN sites are employed, a 5 degree minimum earth station antenna elevation angle is sufficient for good coverage. However, if the terminal at Sioux Falls is used a minimum antenna elevation angle of 2 degrees must be assumed to provide complete

Table D.1.4.2-1 Signal Margin for the 240 Mbps and 20 Mbps Wideband Links (678 km orbit; X-Band)

		240 Mbps Link		20 Mbps Link* Steerable S/C Antenna		Fixed S/C Antenna	
S/C Trans. Power	dBW	6.0	(4W)	0.5**	(4W)	16.0	(40W)
Ckt. Losses	dB	2.5		2.5		1.5	
S/C Antenna Gain	dB	30.0	(1.6')	30.0		5.0	($\pm 50^{\circ}$)
Ant. Pointing Loss	dB	3.0		3.0		-	
S/C ERP	dBW	30.5		25.0**		19.5	
FSL	dB	180.0 (2° elev.)		173.0 (30° elev.)		173.0 (30° elev.)	
O_2/H_2O	dB	1.0		0.2		0.2	
Rain	dB	3.1		0.4		0.4	
Cloud	dB	3.0		0.3		0.3	
Propagation Losses	dB	187.1		173.9		173.9	
Ground Ant. Gain	dB	55.4	(30')	41.5	(6')	46.7	(11')
Pointing Loss	dB	0.5		1.5		1.5	
Surface Tolerance Loss	dB	0.3		0.3		0.3	
Ckt. Loss	dB	0.5		0.5		0.5	
Dual Feed Loss	dB	0.5		-		-	
Net Ant. Gain	dB	53.6		39.2		44.4	
K	$dBW/^{\circ}k/Hz$	-228.6		-228.6		-228.6	
T	$dB^{\circ}k$	22.2		27.8		27.8	
C/kT	dB/Hz	103.4		91.1		90.8	
R	dB/Hz	83.8		73.0		73.0	
E_b/N_o (BER=10^{-5})	dB	12.0		12.0		12.0	
Margin	dB	7.6		6.1		5.8	

----

*For 20 Mbps link assumed 30° minimum elevation of LCGS

**For LCGS, S/C Transmitter power controlled to 25.0 dBW power flux density density limit

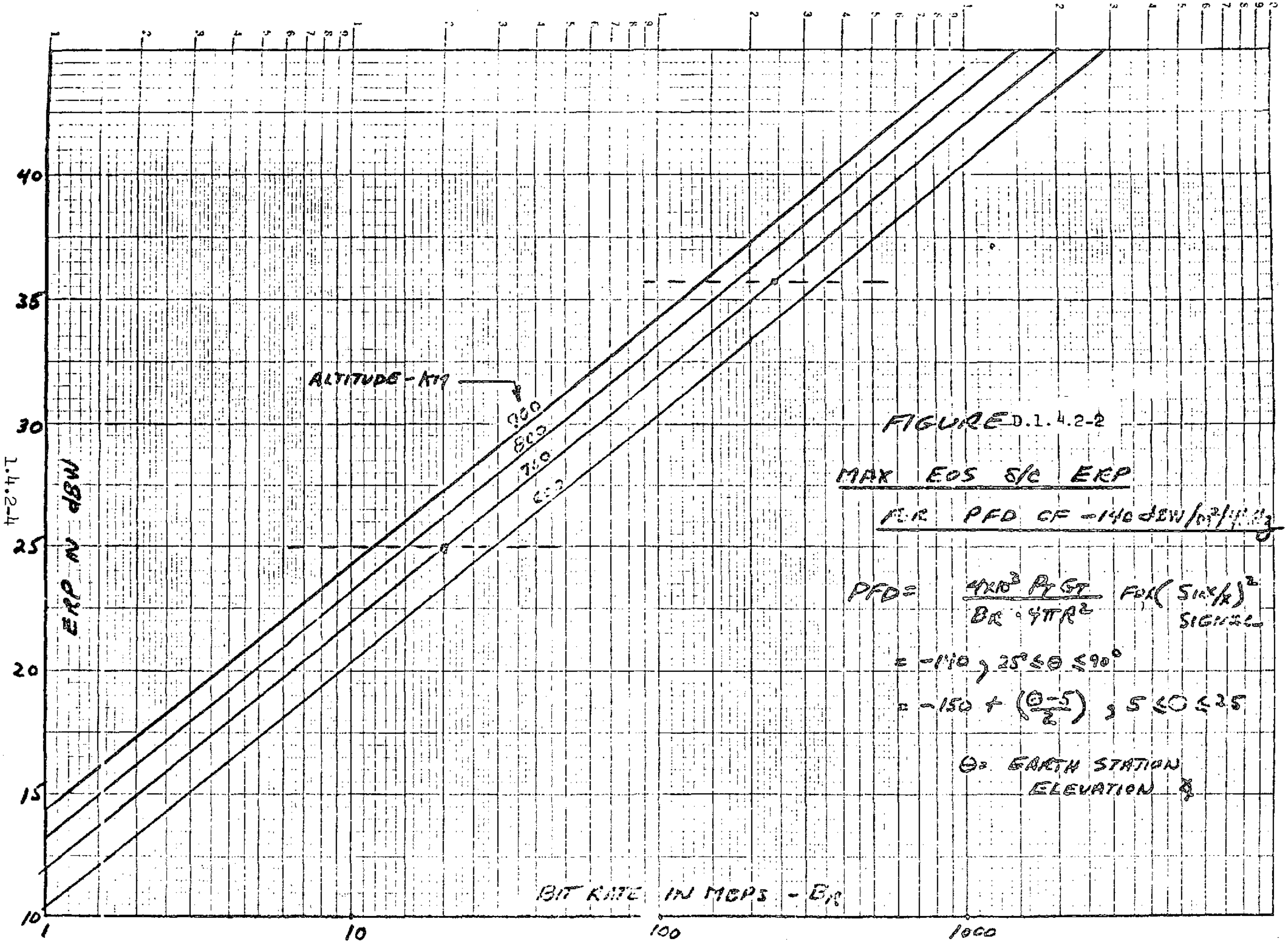
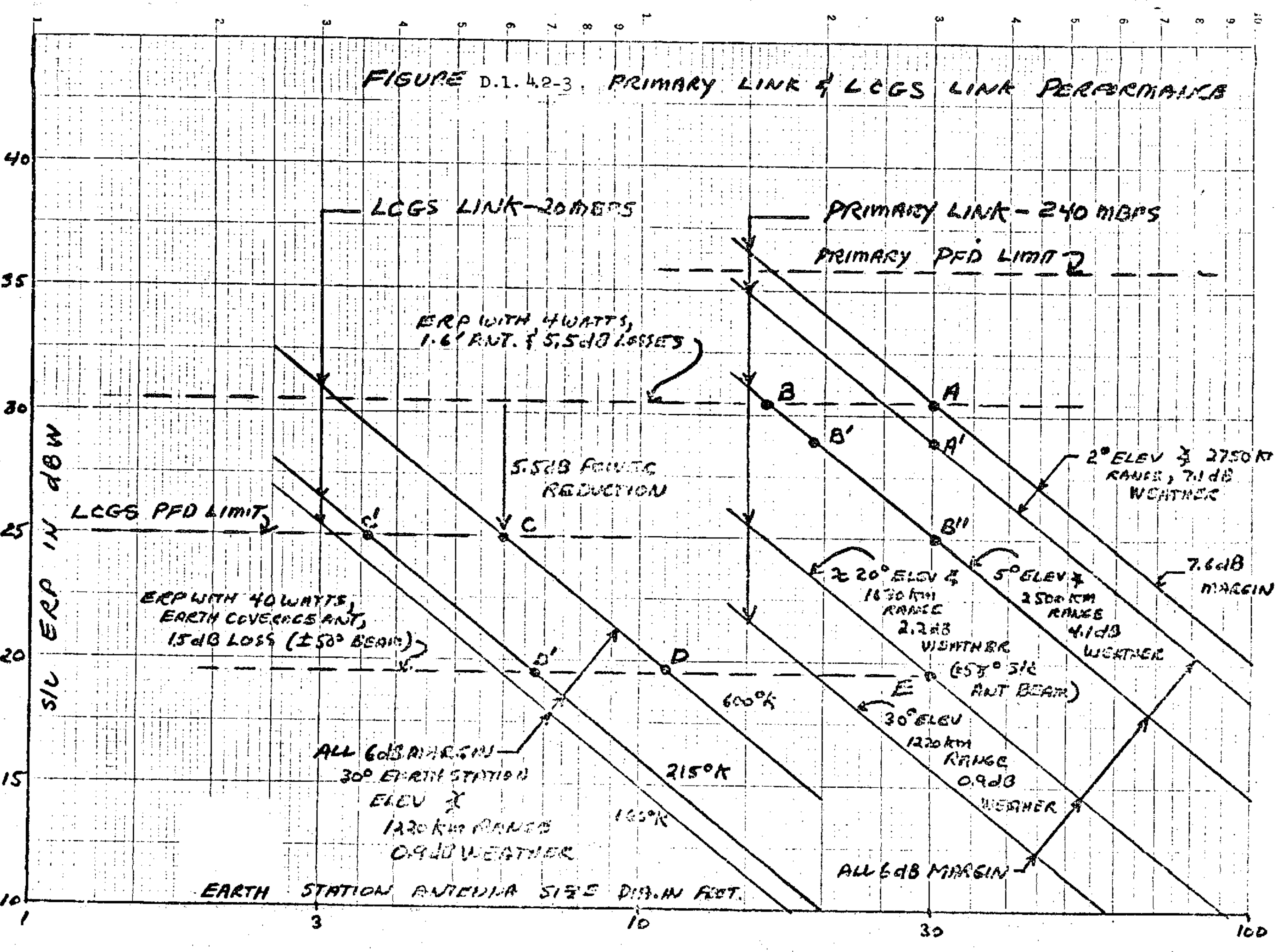


FIGURE D.1.4.2-2

MAX EOS S/C ERP

FOR PFD OF -140 dBW/m²/4 kHz

FIGURE D.1.4.2-3. PRIMARY LINK & LCGS LINK PERFORMANCE

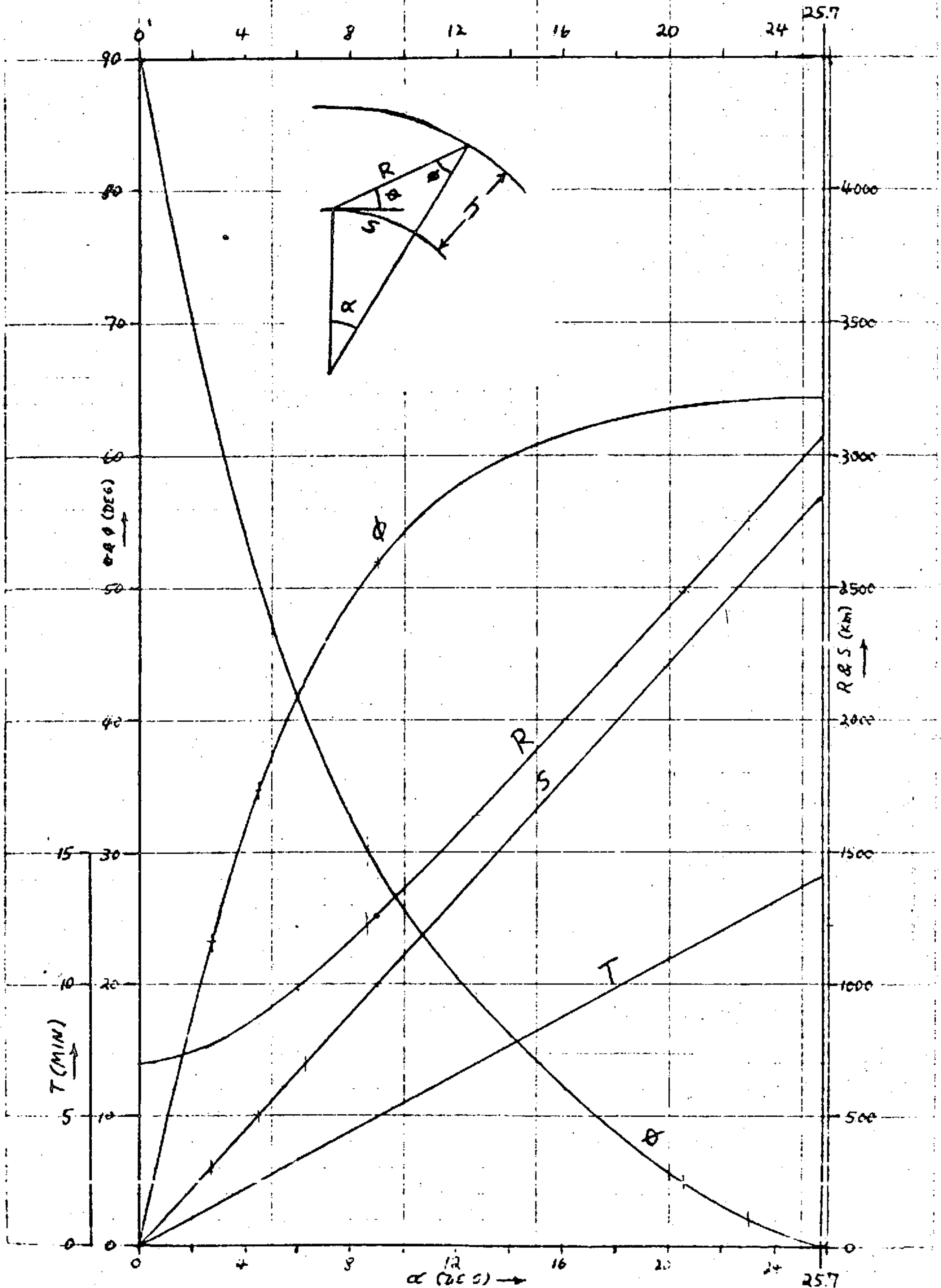


CONUS coverage including Southern Florida. The alternative possibilities for the primary link are shown in Figure D.1.4.2-3. Point A is the power budget design point of Table D.1.4.2-1 resulting in a 7.6 dB margin. The S/C 4 watt power device coupled with a high gain steerable antenna is essential to support this link. If a 6 dB margin is assumed sufficient above the losses of Table D.1.4.2-1 for this link, then an adjustment to Point B is possible, reducing the required ERP by 1.5 dB. This can allow reducing the S/C antenna diameter from 1.6 foot to 1.34 feet. For the STDN earth station scenario the 5 degrees minimum elevation angle reduces free space loss and weather losses as shown on Figure D.1.4.2-3. For this case and the 30.5 dBw S/C ERP and earth station antenna diameter of only 16 feet is required (Point B). If the S/C ERP is reduced by the same 1.5 dB as above, then this terminal size must be 19 feet (Point B'). However, if new STDN antennas are undesirable, then the 30 foot STDN station can be employed (Point B") incurring an additional link margin of 5.5 dB (11.5 dB total) for the 30.5 dBw S/C ERP case.

The basic conclusion from this work is that a high gain steerable S/C antenna dBw's essential to support the primary link for either the present STDN sites or the postulated Sioux Falls site. Additional spacecraft trades can be made to reduce the size of the S/C antenna by increasing the transmitter power. However, the gain of a 1.6 foot parabolodal antenna at X-Band is 30 dB. Increasing the transmitter power 10 dB from 4 watts to 40 watts will require a 20 dB antenna gain which is still a steerable device (16 degree 3 dB beam). Later discussion for the LCGS link will point out the use of the LCGS earth coverage antenna Alternative 2 for supporting the primary link in the event of the 1.6 foot steerable antenna failure.

Figure D.1.4.2-4 is included to document the line of sight (LOS) range and antenna elevation angles employed in the construction of Figure D.1.4.2-3 LCGS Link Discussion.

The LCGS link must support 20 Mbps into a single low cost earth station. Table D.1.4.2-1 has sized two possibilities: one meeting the maximum permissible ITU power flux density (PFD) (Point C of Figure D.1.4.2-3) and one employing a 40 watt power tube and an earth

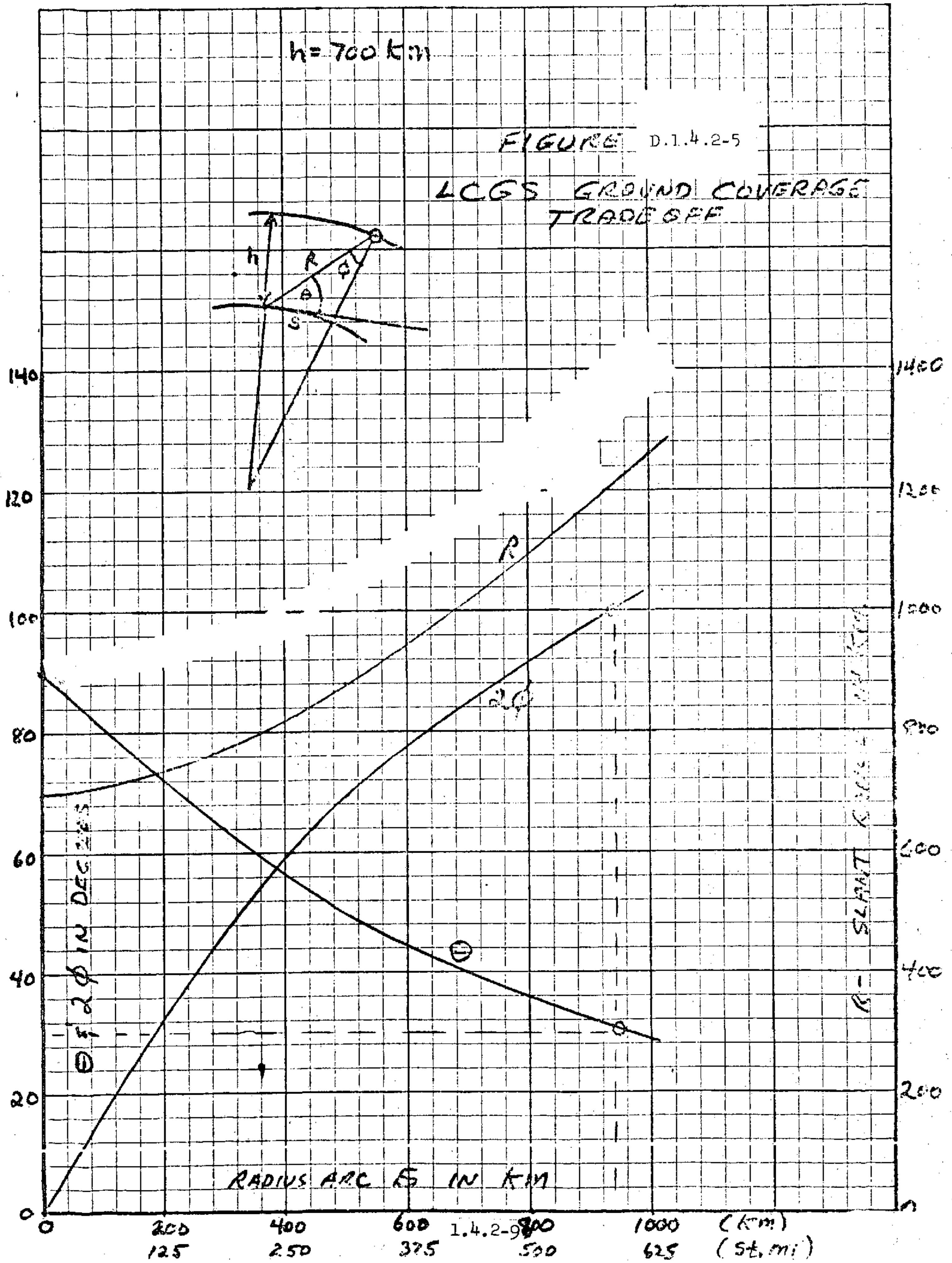


R
S
α
ϕ
θ
T
θ & ϕ (DEG)
R & S (KM)
T (MIN)
α (DEGS)
25.7

coverage fixed S/C antenna (Point D of Figure D.1.4.2-3). Figure D.1.4.2-3 presentation of this data has made slight adjustment to the numbers of Table D.1.4.2-1 to equalize the margin to 6 dB. The results indicate that a 6 foot antenna will support the link if the S/C ERP is maintained at 25 dBw and an 11 foot antenna will be required if the reduced earth coverage ERP case is employed. Both of these cases assume an earth terminal system noise temperature of 600°K (low noise field effect transister). Cases for improvement of system noise temperature by the use of an uncooled parametric amplifier are also shown (Points C' and D') for the above S/C ERP cases.

The 25 dBw ERP case is to employ the 4 watt X-Band power device coupled with the 1.6 foot steerable antenna. However, the output must be adjusted downward to achieve the 25 dBw ERP (an attenuator controlling the drive to the transmitter power device). The reduced earth coverage ERP case does not violate the PFD limitation.

Additional tradeoffs are still being considered for the fixed beam S/C antenna in order to reduce the transmitter power by reducing the ground coverage (lowering the S/C antenna beam angle and correspondingly increasing the earth station antenna minimum elevation angle). The tradeoff in ground coverage (radius arc about the subsatellite point) is depicted in Figure D.1.4.2-5. The present design employs a shaped beam S/C antenna having a 4 dB gain at the ±50 degree points (Figure D.1.4.2-6). Such a design will result in a ground radius arc coverage of ±950 km (±600 st. mi.). By reducing the S/C antenna fixed beamwidth, additional antenna gain can be realized at the expense of reduced ground coverage. The reduced ground coverage also lowers the maximum slant range to the S/C, thereby lowering the free space loss. By reducing the ground coverage from 950 km, the S/C antenna beamwidth goes from 100° to 70°, and the range changes from 1220 km to 880 km. Perhaps an increase of 2 dB can be realized in antenna gain at the ±35° beam edges and a lower free space loss by 2.8 dB resulting in 4.8 dB net gain which can be used to reduce the transmitter power from 40 watts to 13.3 watts. The desirability of this tradeoff is as yet not clear considering the reduced coverage resulting and considering spatial and frequency acquisition times that could be an appreciable portion of the total visibility time. The total visibility time for this reduced coverage example is only 2.5 minutes (see Figure D.1.4.2-4).


h= 700 Km
FIGURE D.1.4.2-5
LCGS GROUND COVERAGE TRADEOFF
h
R
α
θ
S
R
2φ
θ
θ or 2φ IN DEGREES
R - SLANT RANGE IN KM
RADIUS ARC S IN KM
0
20
40
60
80
100
120
140
200
400
600
800
1000
1200
1400
0
200
400
600
800
1000
(Km)
125
250
375
500
625
(St. mi)

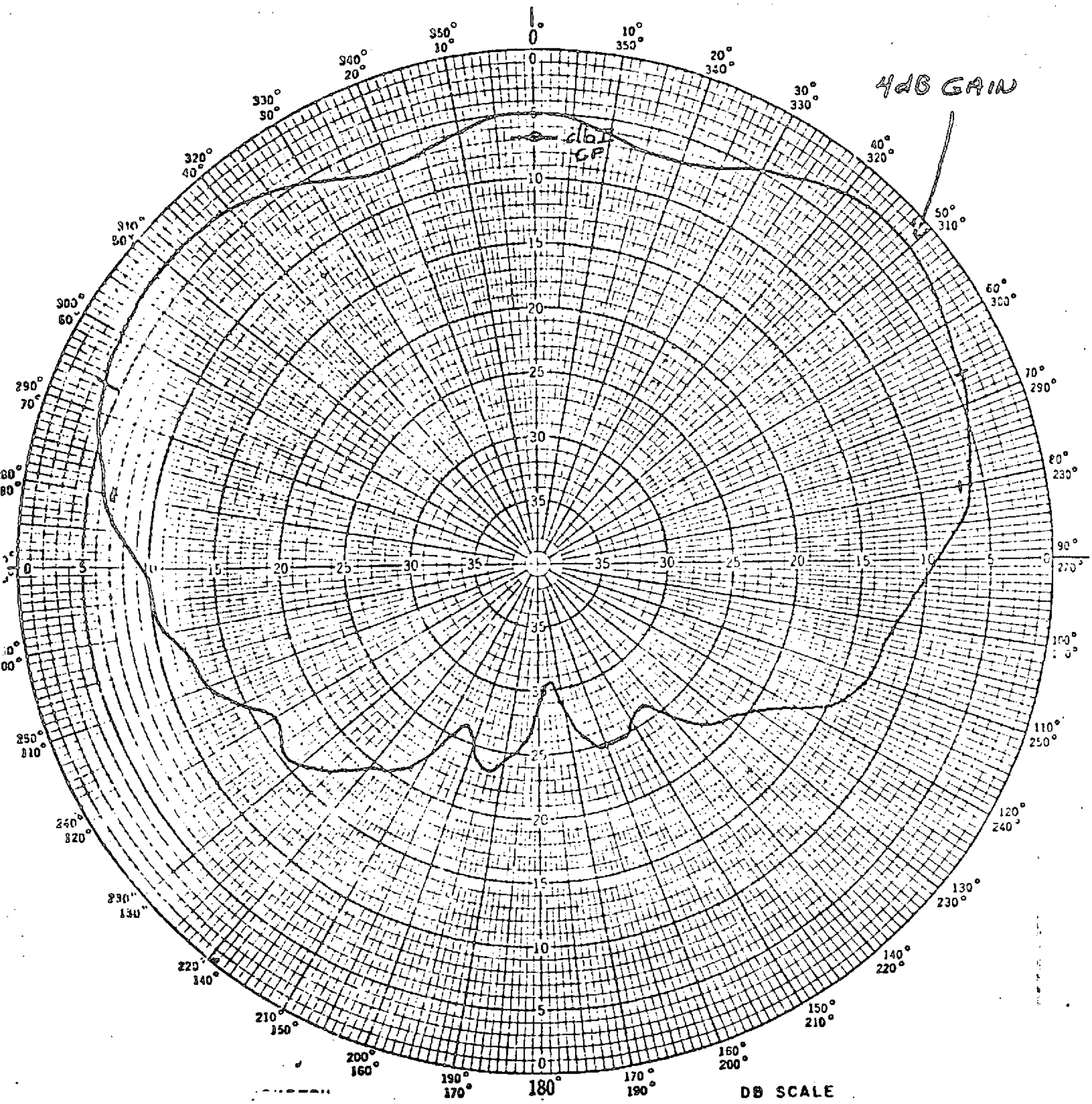


Fig. D.1.4.2-6 Typical Fixed Antenna Pattern with Shaped Beam

A final point to be considered is the use of the earth coverage antenna and power device as a backup for the primary link support. Figure D.1.4.2-3 includes this case (Point E). It has been determined that the 30 foot STDN terminal could receive the 240 Mbps data at an elevation angle of approximately 20 degrees using the 19.5 dBw earth coverage ERP primarily used for LCGS support in Alternative 2. The exact elevation angle is quite sensitive to the assumed weather losses expected at these elevation angles.

Summary

The designs picked for further consideration are summarized as follows.

Alternative 1

Primary Link:

S/C Power - 4 watts

S/C Antenna - 1.6 foot steerable

Earth Station - 30 foot STDN with 5^{o} minimum elevation angle or new 30 foot terminal (Sioux Falls) with 2^{o} minimum elevation angle

Performance Points A and B" on Figure D.1.4.2-3

LCGS Link:

S/C Power - 4 watts reduced to 1.2 watts

S/C Antenna - 1.6 foot steerable

LCGS Terminal - 6 foot with 600^{o}K system temperature with minimum elevation angle of 30^{o}

Performance Point C on Figure D.1.4.2-3

Alternative 2

Primary Link: Same as Alternative 1 with backup capability Point E on Figure D.1.4.2-3

LCGS Link:

S/C Power - 40 watts

S/C Antenna - ±50 degree fixed beam with 4 dB gain at ±50 degrees

LCGS Terminal - 11 foot with 600^{o}K noise temperature

Performance Point D on Figure D.1.4.2-3

D.1.4.2.3 SELECTED CONFIGURATION AND RATIONALE

A cut has been taken at comparing Alternative 1 and Alternative 2 from the viewpoint of size, weight, power consumption, cost and risk. Table D.1.4.2-2 lists the above factors with system impact estimates. Preliminary results indicate that S/C sizing and total cost considerations would favor the fixed beam antenna approach (Alternative 2) and prime power considerations would favor the steerable beam approach. S/C weight, cost and risk impacts are judged at this time to be equal between Alternative 1 and Alternative 2. Reliability considerations, however, will probably tip the scale in favor of Alternative 1, since with this approach, two identical steerable antennas are available to provide redundancy in case of failure.

D.1.4.2-4 SUPPORTING STUDIES

A number of study areas need to be considered for the wideband communications subsystem. Some of these are: frequency plan, modulation choice, acquisition, and technilogy studies - antenna and power sources. To date, the frequency planning study area has been investigated. The result of this work is recorded in the following paragraph.

(1) Frequency Plan for Wideband Communications

The Problem

The X-Band allocation of 8.025 - 8.400 GHz is recommended for the operational EOS wideband communications. This frequency space must be shared by the primary 240 Mbps data rate link and the LCGS 20 Mbps data rate link. The optimum manner of doing this and the resulting total losses to the demodulation theoretical E_b/N_o (energy per bit to noise density ratio) to achieve 1×10^{-5} bit error rate is the issue at hand.

The choice of modulation for the wideband link is clearly QPSK (quatinary phase shift keying) to conserve spectrum at a reasonable demodulator implementation complexity. The spectral bandwidth for QPSK is one half that for BPSK (binary phase shift keying) but the detection E_b/N_o is the same for either QPSK or BPSK (theoretically 9.6 dB for 1×10^{-5} average bit error rate). The choice of modulation for the

Table D.1.4.2-2 Alternative Configuration Comparison

Link Option / System Impact	LCGS Alternatives: Alternative 1 — Steerable Narrow Beam Antenna	Alternative 2 — Fixed Wide Beam Antenna	Impact Evaluation
Weight (lbs)			
Antenna	10.0	1.2	
P.A.	2.0	14.0	
Total	12.0	13.2	Equal
Size			
Antenna	2.5 ft^3 (2300 $in.^3$)	10.0 in^3	Favors Fixed
P.A.	2x2x5=20 $in.^3$	14.5x4.5x6.0=390 $in.^3$	Beam Antenna
Total	2320 $in.^3$	400 $in.^3$	
Power (Prime Watts)			
Antenna	10*	0	Favors Steerable Steerable
P.A.	18	103	Beam Antenna
Total	28	103	
Cost ($)			
S/C Antenna	170K[2]	30K[3]	
S/C P.A.	40K[2]	100K[1]	
Subtotal	210K	130K	Equal
E/T Antenna (100)	10Kea.1000K	20K ea.2000K	Favors Beam Antenna
Total	1210K	2200K	
Risk	Moderate (Steerable Antenna but with Redundancy)	Moderate (Single Thread 40W TWTA)	Equal

Notes: 1 Recurring cost only; tube is developed.
2 Recurring cost only; primary link picks up nonrecurring cost.
3 Includes nonrecurring and recurring.

LCGS link can be BPSK due to the lower rate (spectrum conservation is not so much required) and the desire for low demodulator implementation complexity.

In order to share the spectrum allocation available in an optimum non-interfering manner two approaches were investigated. One of these employs frequency division multiplexing with minimum frequency spacing coupled with individual signal filtering. The other approach places both signals on the same nominal carrier frequency with filtering on each and achieves minimum interference between signals by sending each on separate narrow beam spacecraft antennas. The penalty paid for the first approach is spectrum occupancy and for the second minimum separation requirements for LCGS link and primary link earth stations (and, of course, the constraint that both link S/C antennas are the same size and produce relatively narrow beams at X-band).

Results

The above defined approaches were analyzed employing a previously developed channel simulation model. For the frequency division approach it was first determined that the closest possible spacing that could be achieved was to place the 20 Mbps LCGS signal in the first (sin x)/x spectrum null of the 120 Mega symbol per second (240 Mbps) primary link signal. The resulting losses to theoretical BPSK or QPSK signal performance are depicted in Figures D.1.4.2-7 and D.1.4.2-8 as a function of probability of error. These losses include both the respective adjacent signal interference and filter intersymbol interference effects. The term E_s/N_o has been employed (energy per QPSK or BPSK symbol to noise density ratio) in these and subsequent figures. The relationships between E_s/N_o and E_b/N_o are:

$$E_b/N_o = E_s/N_o \text{ (BPSK)}$$

$$E_b/N_o = E_s/N_o - 3 \text{ dB (QPSK)}$$

Note that all systems comparison is done on the basis of E_b/N_o and a given bit error rate. Also shown in these figures is the effect of the adjacent interferer being larger in power than the desired demodulated signal.

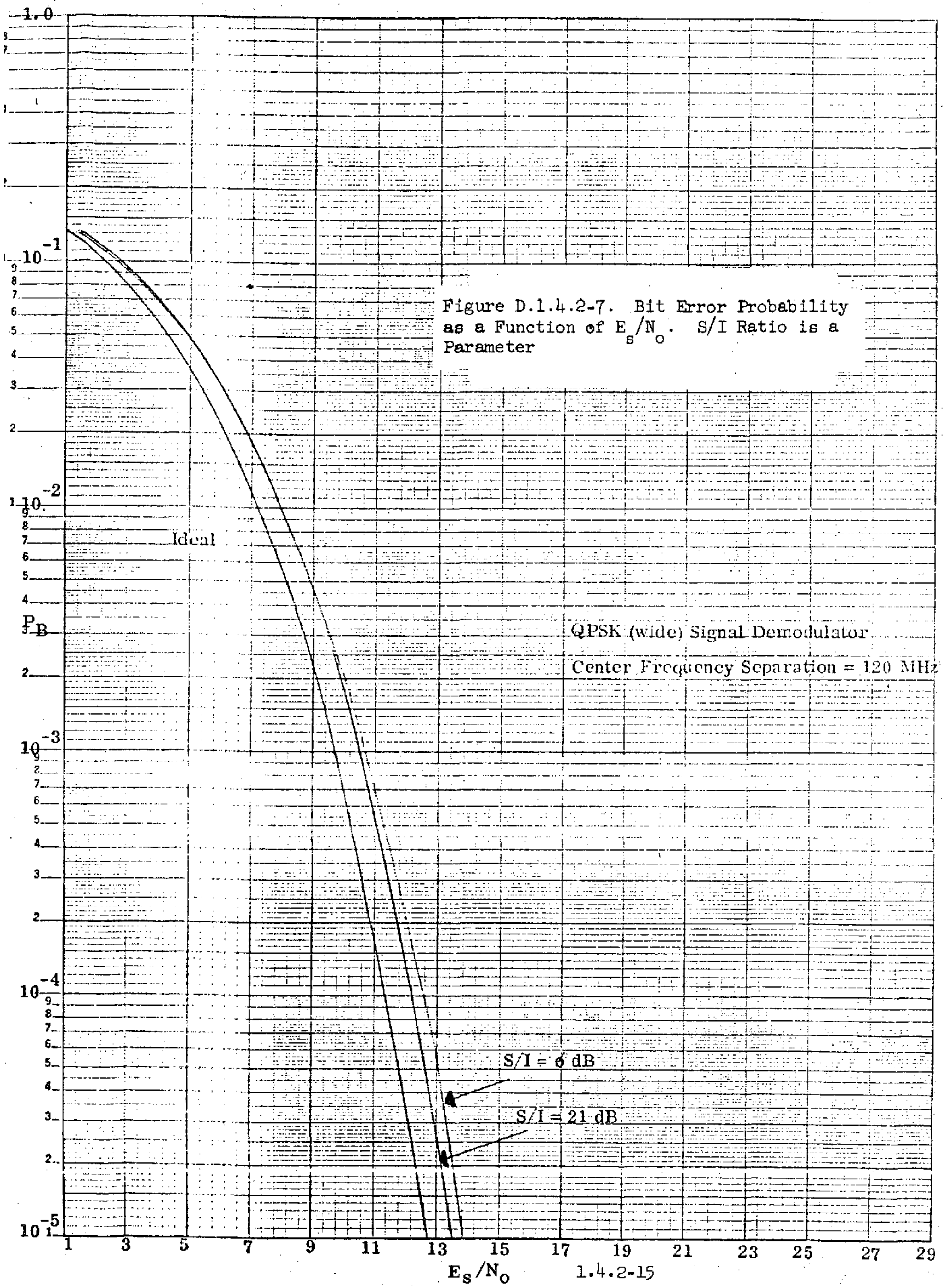


Figure D.1.4.2-7. Bit Error Probability as a Function of E_s/N_o. S/I Ratio is a Parameter

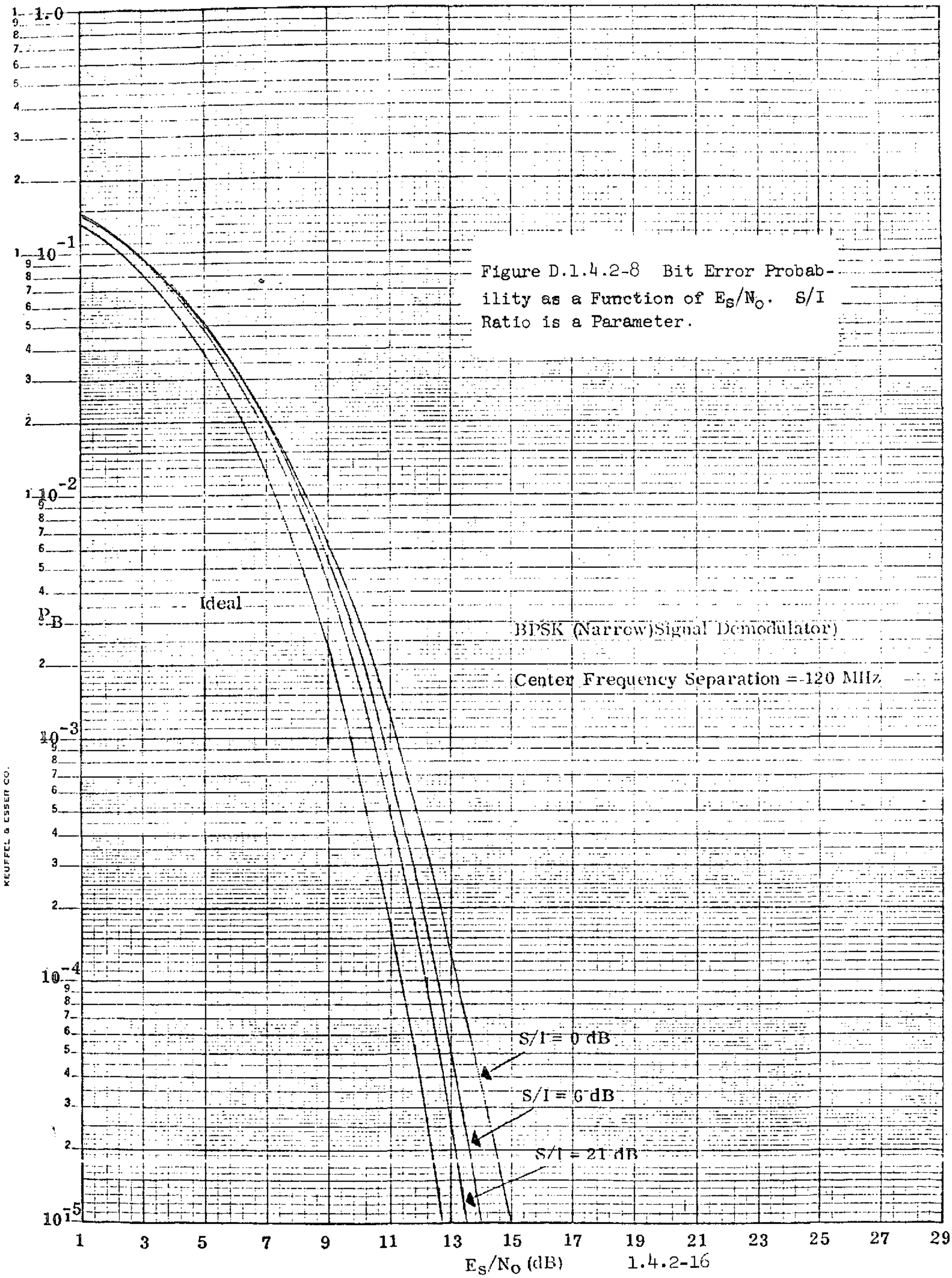


Figure D.1.4.2-8 Bit Error Probability as a Function of E_S/N_O. S/I Ratio is a Parameter.
Ideal
P_B
BPSK (Narrow)Signal Demodulator)
Center Frequency Separation = 120 MHz
S/I = 0 dB
S/I = 6 dB
S/I = 21 dB
1.0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
1
3
5
7
9
11
13
15
17
19
21
23
25
27
29
E_S/N_O (dB)
KEUFFEL & ESSER CO.

Similar results were determined for the case of both signals on the same frequency. These results are shown in Figures D.1.4.2-9 and D.1.4.2-10.

The effect of narrow S/C antenna beam isolation between LCGS and primary links was determined by plotting an ideal antenna pattern curve for a 1.6 ft. diameter parabola as a function of offset angle from boresight. This curve is recorded in Figure D.1.4.2-11

The final resulting modulation losses as a function of both approaches and the separation angle between S/C antennas is shown in Figure D.1.4.2-12. These results show that 2 dB modulation losses can be sustained for the frequency division approach with no isolation from S/C antennas for the common signal frequency case when S/C antenna beams are separated by a minimum of 22 degrees. This latter result corresponds to a minimum LCGS and primary ground station separation of 280 km.

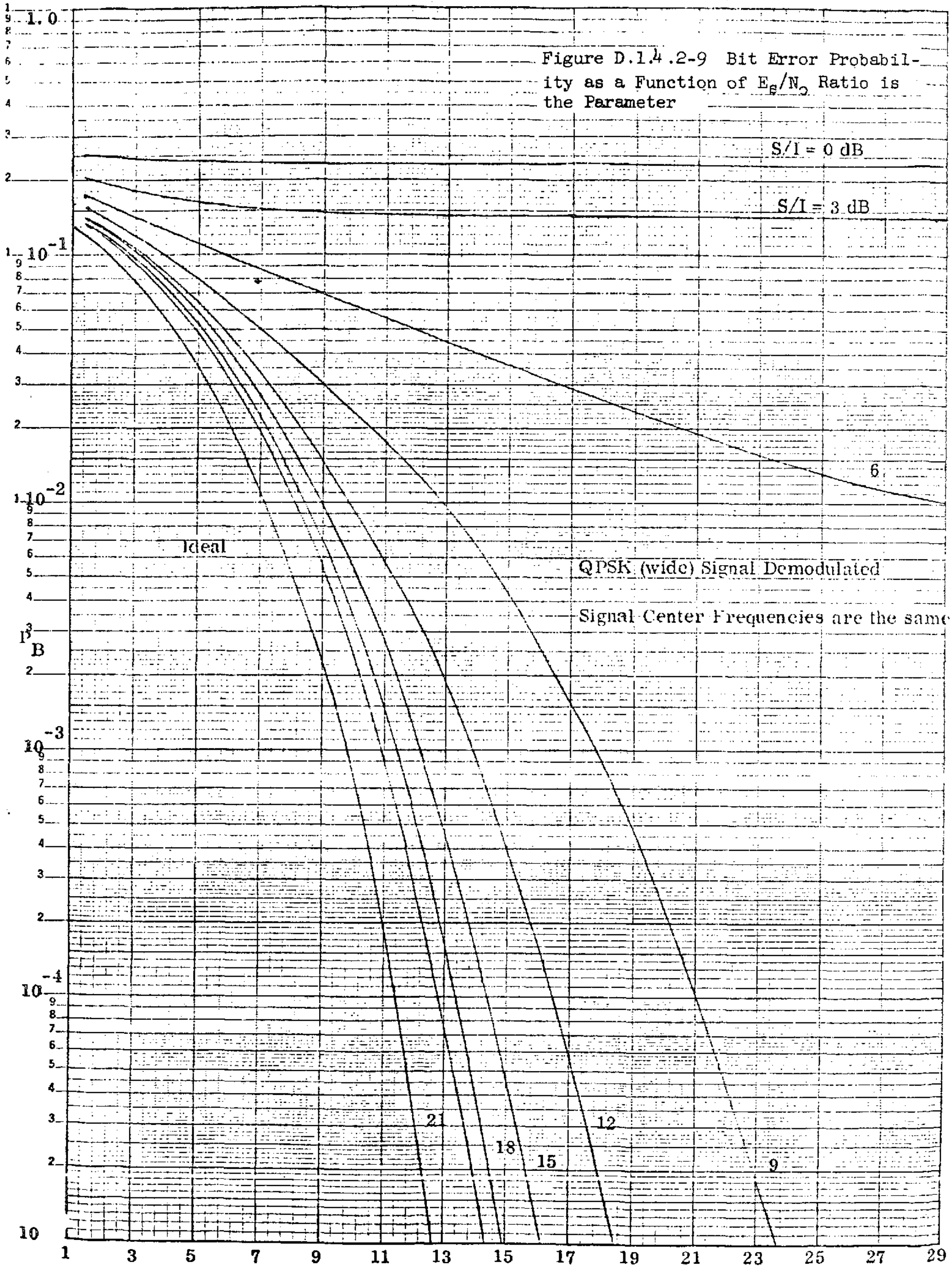


Figure D.1.4.2-9 Bit Error Probability as a Function of E_s/N_o Ratio is the Parameter

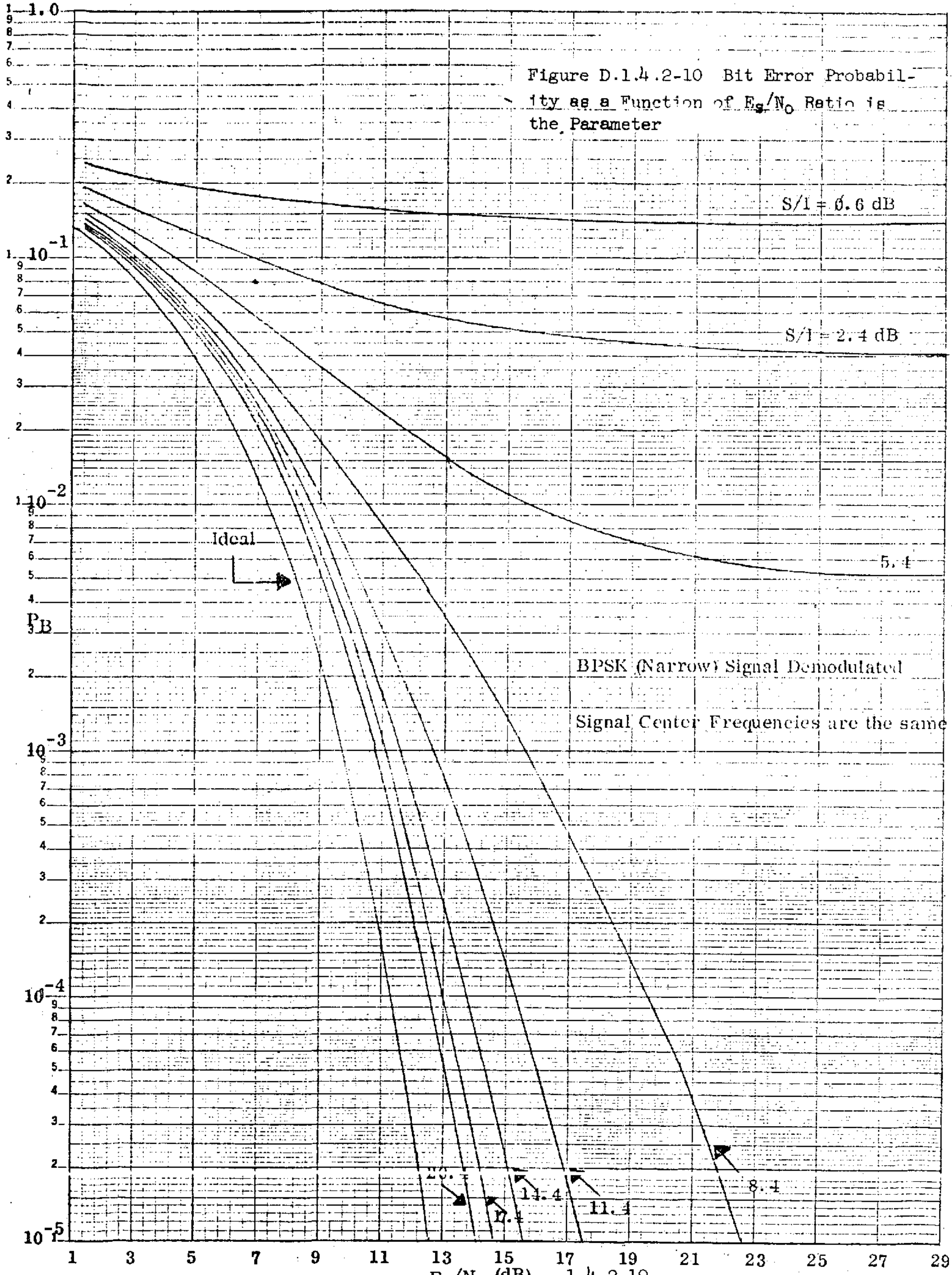


Figure D.1.4.2-10 Bit Error Probability as a Function of E_s/N_0 Ratio is the Parameter
S/I = Ø.6 dB
S/I = 2.4 dB
5.4
Ideal
BPSK (Narrow) Signal Demodulated
Signal Center Frequencies are the same
8.4
11.4
14.4
17.4
P_B
E_s/N_0 (dB)
1.0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
1
3
5
7
9
11
13
15
17
19
21
23
25
27
29

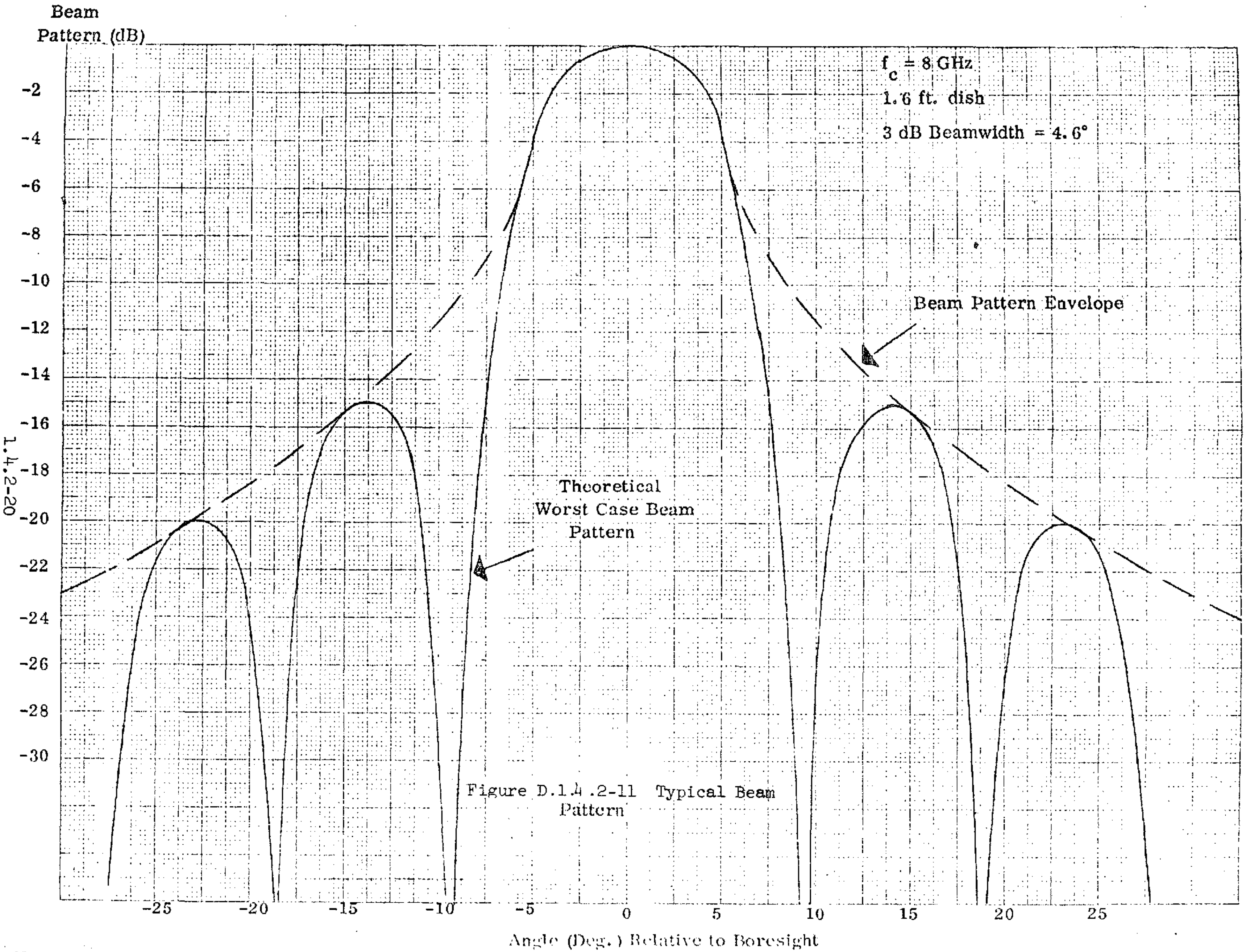


Figure D.1.4.2-11 Typical Beam Pattern

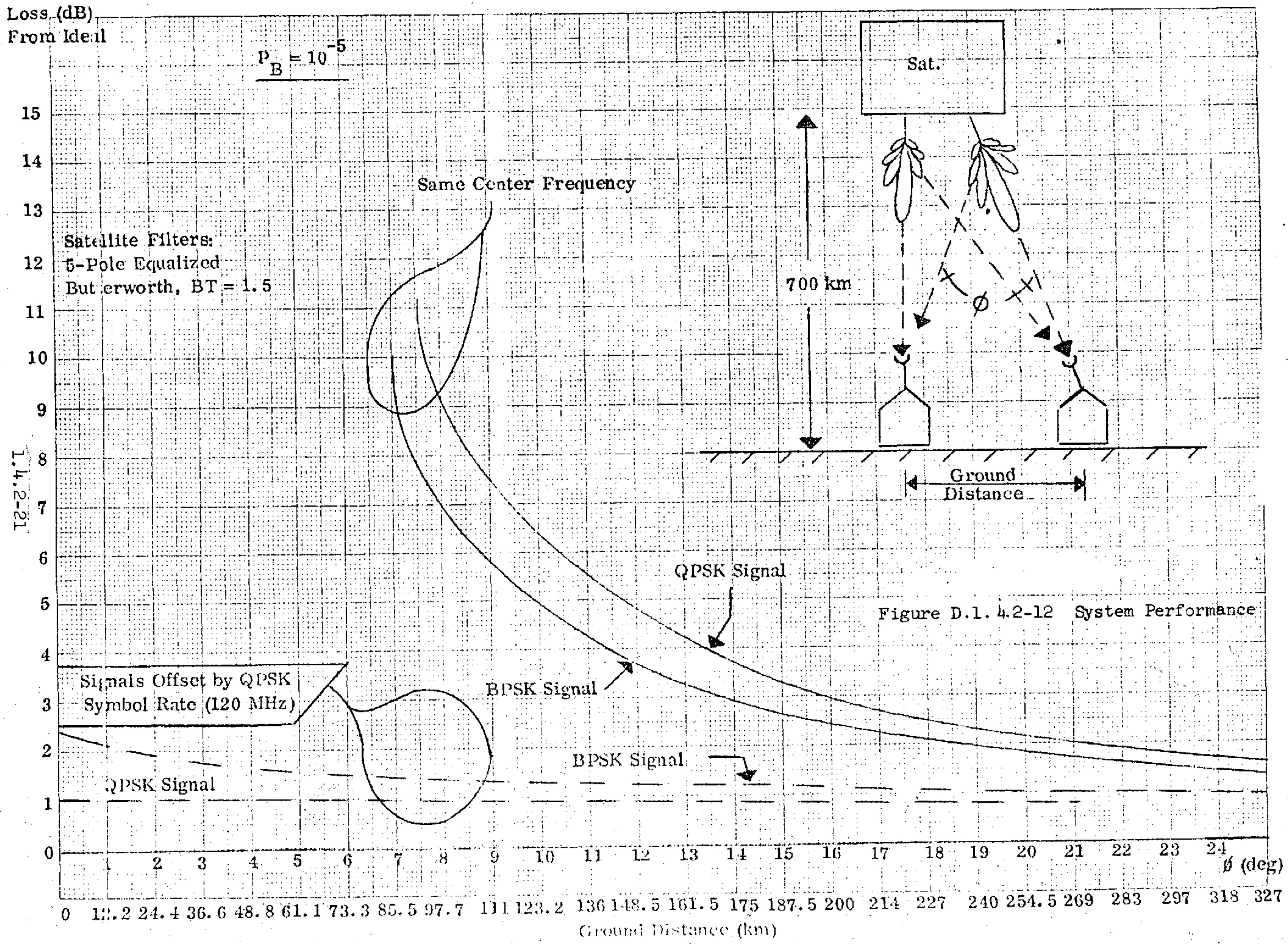


Figure D.1.4.2-12 System Performance

1.4.2-21

1.5 Follow-On Instrument/Mission Accomodations

As of this phase in the study, the review of the accomodations required for the follow-on instruments has been included in section 4.5 of Report #3. The study of this subject is continuing and the results will be included in the study final report.

1.6 Ground Support Equipment

The requirements for GSE depend upon the EOS design and the test program necessary to place the EOS into orbit. For the system definition study it was found that alternate S/C configurations had little impact in the type and quantity of GSE required. The major drivers were the tests to be performed, the time element for test results, the manner in which the test would be conducted and the supporting equipment required for handling the S/C.

Except for instrument operation while installed in the S/C, all instrument GSE was assumed to be provided by the respective instrument vendor.

1.6.1 GSE Requirements

A. S/C Level

(1) Real time analysis of S/C data and operation, less sensor outputs, is required at all test sites. The real time analysis permits instant evaluation of S/C anomalies which may be corrected as test progresses with minimum down time. This results in lower test costs and a shorter test period.

In order to perform real time analysis of the S/C during test, uplink and downlink communications is required. This results in a test stations with an RF front end capability, which has as a result, the added benefit of use during S/C pre-launch and launch test at VAFB, the primary launch site.

(2) All S/C equipment is required to operate during test. Where equipments do not receive interface signals as a result of non-orbital operation, such as the solar array sensor, stimulation is required. The stimulation need not be to orbital levels in all cases, but enough to provide an output sufficient to verify equipment operation. Provision may also be made in the software program of the test station to account for the reduced output for non-orbital level stimulation.

(3) Real time analysis of vehicle performance during thermal vacuum tests of the S/C is required. This analysis will not require any addition to the test station which will be used for nominal S/C testing. Interface cabling with a thermal vacuum chamber will be required and constitutes additional equipment.

(4) During vibration and thermal vacuum tests, additional instrumentation will be added to the S/C (i.e., thermal and vibration sensors) These signals are not part of the normal downlink housekeeping data and will require monitoring during the test.

(5) S/C power will be on during testing. In order to conserve flight batteries (lower cots), ground power, simulating the flight battery characteristics, must be provided for the S/C. The capability must also be provided to simulate the S/C load profile.

(6) Prior to installation of the flight batteries they must be conditioned by discharge and charge cycles.

(7) Assembly of the S/C will be in the vertical position, but its movement inter-and intra-site will be in the horizontal position. Assembly will take place in a 10,000 Clean Room.

Transportation dollies, handling equipment and work stands must accommodate this requirement. In addition, environmental protection must be provided during S/C movement.

(8) All S/C pyro signals, in response to pyro initiation commands must be simulated during test.

(9) S/C tanks require actual fluid or pneumatics, or when not practical, a simulated but inert substitute, to the same cleanliness level required for flight. Handling of such fluids should not cause its contamination or contamination to S/C tanks and lines.

B. S/C Module Level

With a S/S modular approach to S/C assembly, bench C/O of the modules are now required in order to provide for the integration of the equipment within a module. This provides for an overall cost reduction by removing a large portion of S/C checkout from the period of time when S/C integration delay is costly. Each module now is complete and operable in itself, given S/C interface simulation, including appropriate power profiles.

The module checkout benches should be provided with sufficient flexibility so that they can be used as module maintenance benches during the Shuttle operational period.

1.6.2 Equipment Identification

A review and analysis of the test trade study report summary combined with the top level GSE requirements of 4.7-1 has resulted in the identification of the GSE for EOS, to a sufficient depth to permit its costing. These equipments have been divided into three categories, electrical, mechanical and fluid. A review of the test schedule revealed no interference between S/C production and test. As a result, only one of a GSE end item is required. The items with an asterisk have been identified as candidates for added quantities in the event a follow mission scenario calls for more than one launch per year.

A. Electrical Equipment

(1) Test & Integration Station*
(2) Breakout Box Set *
(3) Battery Conditioner *
(4) Test Battery Set *
(5) S/C Power Set & Cables *
(6) Ranging Test Assembly
(7) Pyro Test Set
(8) Interface Cable Set
(9) Solar Simulator
(10) Umbilical Simulator
(11) DITMCO - Program & Cable Set
(12) Power Module C/O Bench
(13) C & DH Module C/O Bench
(14) ACS Module C/O Bench
(15) Propulsion C/O Bench (RCS)
(16) DCS Simulator
(17) S/C Monitor & Control

B. Mechanical

(1) Interface Adapter Set
(2) Hoist Bar & Sling Set *
(3) Support Dolly Verticle - S/C *
(4) Support Dolloy-Horizontal-S/C *
(5) Access Work Stand - S/C *
(6) Skin Storage Rack *
(7) Wt. & C.G. Fixture - S/C
(8) Mass Simulator Set
(9) Support Dolly Low Height - S/C Modules
(10) Shipping Containers S/C Modules
(11) S/C Cover Set *
(12) Humidity Control Kit
(13) Shipping Container - Solar Array
(14) Solar Array Inst. & Deploy. Fixture
(15) Transporters
(16) Indicating Accelerometer Kit
(17) Pyro Inst. Tool Kit
(18) Stage Motor Inst. Fixture
(19) EOS/Shuttle Simulator Adapter
(20) Tie Down Kit Set
(21) Battery Shipping Container
(22) Battery Installation Tool

C. Fluid

(1) GN_2 Conditioning Unit *
(2) GN_2 Regulation Unit *
(3) Volumetric Leak Detector
(4) Mass Spectrometer Leak Detector
(5) RCS Vacuum Test Cart
(6) Fluid Dist. Sys. GAC
(7) Pressure Maintenance Unit *
(8) GN_2 Storage Sys. Transporter
(9) GN_2 Manifold and Supply Platform
(10) Fluid Dist. Sys. Launch Site
(11) Propellant Trans. Assembly *

1.6.3 GSE vs. S/C Activities

A review was made to determine what GSE would be required during each spacecraft activity from assembly to launch. The results are presented in Figure 1.6-1.

The figure also reveals the multiple use being made of the GSE which is moved with the vehicle.

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OF POOR QUALITY

Columns 1–11: DEVELOPMENT & QUAL; columns 12–24: FLIGHT S/C

1. MODULE MASS REP. INSTALLATION
2. MODULE STRUCT. DEV. & ACOUSTIC QUAL
3. MODULE THERMAL VACUUM DEVEL. TEST
4. S/C MODULE & INST. MASS REP. INSTAL.
5. SEPARATION & DEPLOYMENT QUAL TEST
6. MODULE WIRING & COMPONENT INTEG.
7. INSTR. INTEGR. & OBS FUNCT. TEST
8. ACOUSTIC VIB. & PYRO SHOCK
9. THERMAL VACUUM & EMC
10. POST T/V S-S. TESTS
11. SOFTWARE/HARDWARE INTEGRATION
12. MODULE COMPONENT INSTAL. & INTEGR.
13. MODULE ENVIRONMENTAL ACCEPT. TEST
14. POST ENVIRON. TEST MODULE C/O
15. S/C STRUCTURAL FAB & ASSY.
16. S/C WIRING & C/O
17. S/C MODULE INTEGR. & FUNCT C/O
18. INSTR. INTEG. & C/O, IMP & OAS INTEG.
19. OBS SYS. TEST & EMC
20. SEPARATION & DEPLOYMENT MECH C/O
21. OBS SPECIAL TEST
22. WORKMANSHIP ACOUSTIC
23. PACK & SHIP
24. PRE & LAUNCH OPERATIONS

ELECTRICAL EQUIPMENT

Equipment	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
1. TEST & INTEGRATION STATION							•	•	•	•	•						•	•	•		•	•		•
2. BREAKOUT BOX SET						•	•	•	•	•	•	•	•			•	•	•	•		•			•
3. BATTERY CONDITIONER																			•					•
4. TEST BATTERY SET							•	•	•	•	•						•	•	•	•	•	•		•
5. S/C POWER SET & CABLES							•	•	•	•	•					•	•	•	•	•	•	•		•
6. RANGING TEST ASSEMBLY																	•							•
7. PYRO TEST SET								•											•					
8. INTERFACE CABLE SET							•										•	•						
9. SOLAR SIMULATOR							•											•	•		•			
10. INSTRUMENT INTERFACE SIMULATOR							•										•	•						
11. UMBILICAL SIMULATOR																			•					
12. DITMICO- PROGRAM & CABLE SET															•									
13. POWER MODULE C/O BENCH						•						•	•	•			•							
14. C & DH MODULE C/O BENCH						•						•	•	•			•							
15. ACS MODULE C/O BENCH						•						•	•	•			•							
16. PROPULSION C/O BENCH (RCS)						•						•	•	•			•							
17. DCS SIMULATOR							•	•	•	•	•						•	•	•		•	•		•
18. S/C MONITOR & CONTROL							•	•	•	•	•						•	•	•		•	•		•

MECHANICAL EQUIPMENT

Equipment	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
1. INTERFACE ADAPTER SET				•	•	•	•	•	•	•	•				•	•	•	•	•	•	•	•	•	•
2. HOIST BAR & SLING SET	•			•	•		•	•	•	•			•	•	•		•	•			•	•	•	•
3. SUPPORT DOLLY VERTICLE - S/C				•						•						•	•							
4. SUPPORT DOLLY - HORIZONTAL - S/C								•	•															
5. ACCESS WORK STAND - S/C				•			•								•									
6. SKIN STORAGE RACK															•	•	•	•	•					
7. WT. & C.G. FIXTURE - S/C				•															•					
8. MASS SIMULATOR SET	•			•																				
9. SUPPORT DOLLY - LOW HEIGHT - S/C MODULES		•	•										•	•										
10. SHIPPING CONTAINERS S/C MODULES																							•	
11. S/C COVER SET											•												•	
12. HUMIDITY CONTROL KIT																							•	
13. SHIPPING CONTAINER - SOLAR ARRAY																							•	
14. SOLAR ARRAY INST. & DEPLOY. FIXTURE					•															•				
15. TRANSPORTER																							•	
16. INDICATING ACCELEROMETER KIT																							•	
17. PYRO INST. TOOL KIT								•									•							
18. STAGE MOTOR INST. FIXTURE																	•							
19. EOS/SHUTTLE SIMULATOR ADAPTER																				•				
20. TIE DOWN KIT SET																							•	
21. BATTERY SHIPPING CONTAINER																							•	
22. BATTERY INSTALLATION TOOL																	•							

FLUID EQUIPMENT

Equipment	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24
1. GN_2 CONDITIONING UNIT							•	•	•	•							•	•	•					•
2. GN_2 REGULATION UNIT							•	•	•	•							•	•	•					•
3. VOLUMETRIC LEAK DETECTOR							•	•	•	•							•	•	•					•
4. RCS VACUUM TEST CART							•	•		•							•	•	•					
5. FLUID DIST. SYS. , GAC							•	•									•		•					
6. PRESSURE MAINTENANCE UNIT							•	•	•								•		•					
7. GN_2 STORAGE SYS. TRANSPORTER																							•	
8. GN_2 MANIFOLD & SUPPLY PLATFORM																							•	
9. FLUID DIST. SYS. , LAUNCH SITE																							•	
10. PROPELLANT TRANS. ASSEMBLY																								•
11. MASS SPECTROMETER LEAK DETECTOR							•			•							•	•	•				•	

FIGURE 1.6-1

GROUND SUPPORT
EQUIPMENT